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DIVISION San Diego

REPORT ZP-48-35004

MODEL _____

DATE 7 January 1949

TITLE

NOTES ON LONG RANGE

MISSILE DESIGN

SUBMITTED UNDER

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INTRODUCTION AND CONTENTS

The present report is intended as a supplement to CVAC Report No. 48-35003 entitled: "Proposed Development Program on Rocket Type Missiles".

Some of the data of the above report are amplified or substantiated. In addition some further data are submitted underlying Convair's belief that the rocket engine is the power plant best suited for long-range bombardment missiles.

The topics discussed are:

1. Rockets vs Ramjets. A comparison based on RAND studies.page 1
2. Comments on the State of Development of Ramjets for Long Range Missilespage 7
3. Boosters for Long Range Ramjetspage 14
4. Proposed Method for Improving High Altitude Performance of R.M.I. 20,000 Pound Thrust Rocketpage 16
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6. Rocket Missile of 2000 Mile Range with 6000 Pound Glide Type Warhead.page 24

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ROCKETS VS RAMJETS

A COMPARISON BASED ON RAND STUDIES

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ROCKETS VS RAMJETS

A COMPARISON BASED ON RAND STUDIES

~~As a supplement to previous comparisons of Rockets with other type missiles submitted in CVAC Report ZP-48-35003, ZP-6002-002, and "Convair Rocket Missiles",~~ A weight and economic comparison of Rockets vs Ramjets is presented herein based entirely on studies conducted under Project Rand.

It was assumed that identical propellants (Hydrazine-Oxygen) were used in the rocket missile and in the ramjet booster. Any future improvement in rocket propellants would be reflected equally in both types of missile.

Rather than using the present day diffusers the ramjet weights were based on the use of the "reflected wave" type diffuser, which according to RAND is "believed to be attainable in about five years". (Report RA-15072 page 21)

The gross weights vs range plotted in Figure 1 were taken from RAND plots: Figure 32A of Report RA 15063 for the rockets and Figure 211 of Report RA 15070 for the ramjets.

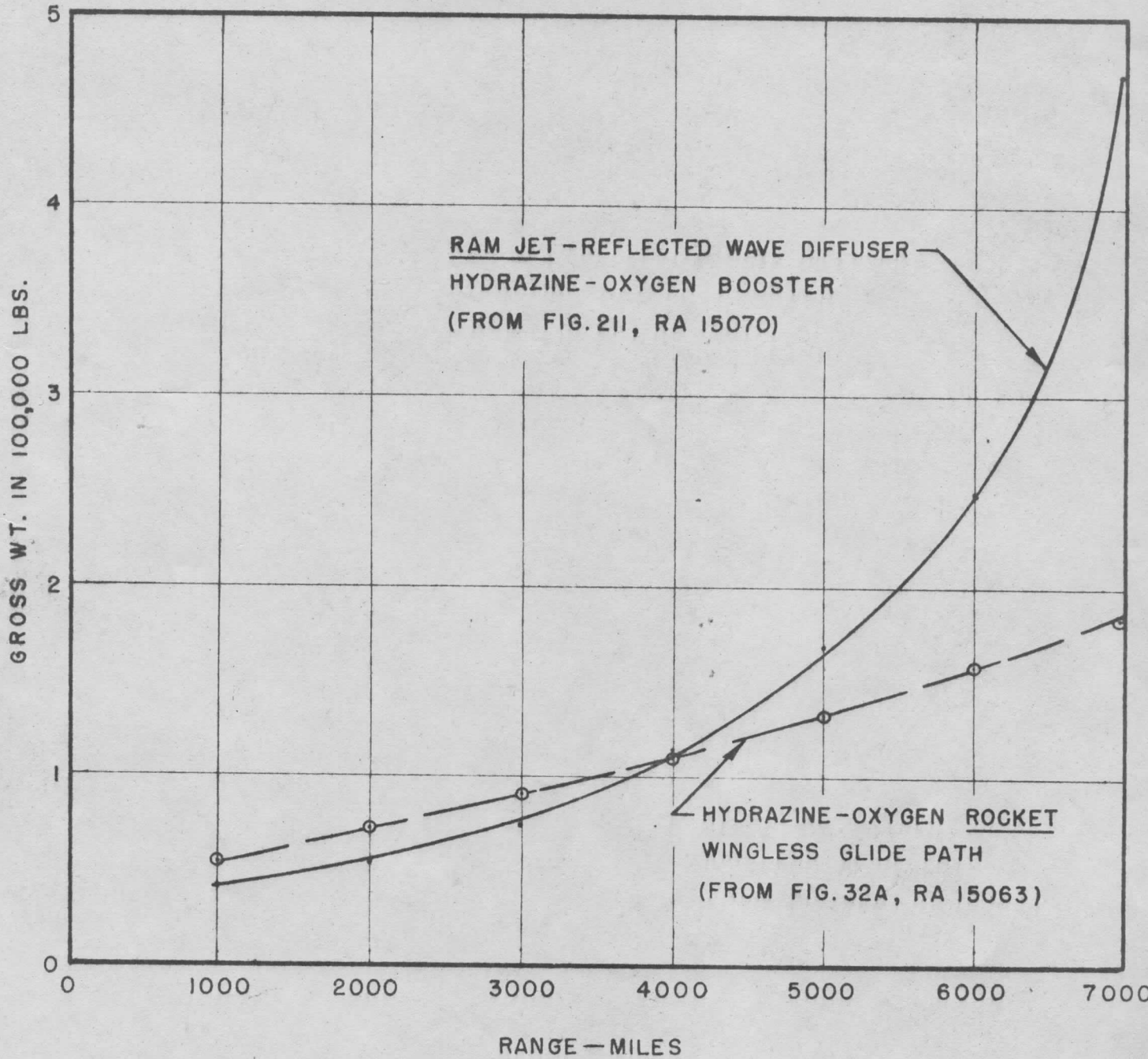
In addition to the weight comparison a cost comparison was made for missiles of 3,000 and 5,000 miles range. The results are shown in Figures 2 and 3.

The costs per pound used in these comparisons for airframe, power plant and guidance are based on CVAC experience to date in the guided missile field. The propellant costs represent current prices in car-load lots with the exception of Hydrazine which is based on Project Rand estimates.

For lack of specific data the cost per pound of the ramjet power plant was assumed the same as that for airframe, even though several of the power plant components - such as valves, pumps, injectors - are of the same degree of complication as those of the rocket power plant.

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GROSS WEIGHT COMPARISON OF
ROCKETS AND RAM JETS
(10,000 LB. WARHEAD)



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5,000 MILE ROCKET-RAMJET ECONOMIC COMPARISON

2-STAGE HYDRAZINE-OXYGEN ROCKET VS REFLECTED WAVE RAMJET WITH HYDRAZINE-OXYGEN BOOSTER
 5,000 MILE RANGE - 10,000 POUND WARHEAD

GO 2411 - VELLUM 357

	Unit Cost	Ramjet Missile		Booster		Ramjet+Booster		Rocket Missile	
	\$/lb	Weight	Cost	Weight	Cost	Weight	Cost	Weight	Cost
Airframe	\$ 27.00	11,609	\$314,000	9,560	\$258,000	21,169	\$ 572,000	12,892	\$348,000
Rocket Power Plant	80.00	----	----	6,010	481,000	6,010	481,000	6,400	512,000
Ramjet Power Plant	27.00	3,460	93,400	----	----	3,460	93,400	----	----
Guidance	100.00	850	85,000	----	----	850	85,000	850	85,000
Payload	----	10,000	----	----	----	10,000	----	10,000	----
Total Wt. Empty	----	25,919	\$492,400	15,570	\$739,000	41,489	\$1,231,400	30,142	\$945,000
Oxygen	.04	----	----	40,300	\$ 1,600	40,300	\$ 1,600	54,858	\$ 2,190
Hydrazine	.45	----	----	44,130	19,900	44,130	19,900	60,000	27,000
Gasoline	.025	39,081	\$ 980	----	----	39,081	980	----	----
Total Fuel	----	39,081	\$ 980	84,430	\$ 21,500	123,511	\$ 22,480	114,858	\$ 29,190
TOTAL	----	65,000	\$493,380	100,000	\$760,500	165,000	\$1,253,880	145,000	\$974,190

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FIGURE 2

FIGURE 3

3000 MILE ROCKET-RAMJET ECONOMIC COMPARISON

2-Stage Hydrazine-Oxygen Rocket vs. Reflected Wave
 Ramjet with Hydrazine-Oxygen Booster

3000 Mile Range - 10,000 Pound Warhead

	UNIT COST	RAMJET MISSILE		BOOSTER		RAMJET + BOOSTER		ROCKET	
	\$/Lb	Wt.	Cost	Wt.	Cost	Wt.	Cost	Wt.	Cost
Airframe	\$ 27.00	4445	\$120,000	4400	\$119,000	8845	\$239,000	7160	\$193,000
Rocket Power Plant	80.00	---	---	3620	289,000	3620	289,000	5218	418,000
Ramjet Power Plant	27.00	1600	43,400	---	---	1605	43,400	---	---
Guidance	100.00	850	85,000	---	---	850	85,000	850	85,000
Payload	---	10000	---	---	---	10000	---	10000	---
Total Wt Empty	---	16900	\$248,400	8020	\$408,000	24920	\$656,400	23228	\$696,000
Oxygen	.04	---	---	19000	\$ 760	19000	\$ 760	32400	\$ 1,300
Hydrazine	.45	---	---	20980	9,450	20980	9,450	35672	16,100
Gasoline	.025	13100	\$ 327	---	---	13100	\$ 327	---	---
Total Fuel		13100	\$ 327	39980	\$ 10,210	53080	\$110,537	68072	\$ 17,400
Total		30000	\$248,727	48000	\$418,210	78000	\$666,937	91300	\$713,400

FIGURE 3

CONCLUSION:

The above comparisons indicate that for ranges up to 4000 miles the Hydraxine-Oxygen Rocket and the ramjet with "Reflected Wave" dif-fuser and Hydrazine-Oxygen booster are closely comparable as to gross weights. Above 4000 miles the ramjet weight rapidly becomes exces-sive.

The economic comparison for 5000 mile range missiles indicates the cost of the ramjet missile to be 29 percent greater than the cost of the rocket missile. At a 3000 mile range the rocket missile cost exceeds the ramjet cost by 7 percent. Judging from the shape of the curves in Figure 1 this differential should be approximately the same at 2000 mile range.

It is interesting to compare the RAND values with values attain-able in the not so distant future. The CVAC glide rocket described in this report and the MX-770 ramjet were selected to this end. In-asmuch as these missiles have different ranges and different payloads they were compared on the basis of cost per ton mile of payload.

	MISSILE		REFERENCE		COST IN \$/TON MILE
GLIDE ROCKET	RAND	5000 mile	page	4	39.0
	RAND	3000 mile	page	5	47.5
	CVAC	2000 mile	page	45	54.5
RAMJET	RAND	5000 mile	page	4	50.1
	RAND	3000 mile	page - 35003	p. 36	244.4
	MX-770	1000 mile	ZP-48-35003	p. 36	261.3

These figures emphasize the fact that today the rockets are much closer than the ramjets to the ultimate perfection envisioned by RAND.

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COMMENTS ON THE STATE OF
DEVELOPMENT OF RAMJETS FOR
LONG RANGE MISSILES

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I. INTRODUCTION:

The selection of rocket propelled missiles over Ramjets for long range ground-to-ground application was based on considered study of the two methods of propulsion. As one of the principal Contractors in the Navy Bureau of Ordnance's Project Bumblebee, with responsibilities for missile fabrication and flight test, laboratory operation, and combustion research, Convair is peculiarly qualified to discuss the question of Ramjets. Coupled with two years of ramjet testing experience both at Convair operated Ordnance Aerophysics Laboratory at Daingerfield and the San Diego facility, Convair is fortunate in having developed and operated rocket power plants as part of MX-813 and MX-774 programs. This combined experience is valuable in selecting the most suitable propulsion method for long-range ground-to-ground missiles.

Convair's choice of rockets for the long-range bombardment type missiles, in addition to other considerations, is based on sound knowledge of the status of development and the possibilities of various propulsion methods. A brief review of the ramjet art is presented to review the thinking that went into this selection. However, it must be emphasized that the conclusions drawn are based on the requirements for long-range bombardment missiles only, and are not necessarily true for all ramjet applications. Ramjets can be shown to be the optimum power plants for other type missiles.

II. SUMMARY AND CONCLUSION:

The following comments are intended to show that ramjets in their present stage of development are unsuitable for long range propulsion, and that many of the problems which have been indicated by the Bumblebee activity will remain serious obstacles even after a solution for the anti-aircraft ramjet weapon has been reached.

1. Flight tests have confirmed the laboratory findings most satisfactorily. They seem to indicate that present articles will not operate in their design Mach number range at altitudes significantly higher than 35,000 feet.
2. Experience with the scaling problem clearly indicates the need for additional knowledge before extrapolations to arbitrary size are possible. ^{40,000} Therefore, test facilities not yet under construction are a prerequisite to the development of large long-range ramjet missiles.
3. A failure of certain types of ramjet combustors to achieve ignition at launching Mach numbers has been discovered. Solutions to this problem may place some limitations on the launching configuration of some missiles.

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- They don't get hot at 100%*
4. While methods are under development for cooling the hot elements of the long-range ramjet, experience has shown that some loss in efficiency will almost certainly be sustained to accomplish such cooling.
 5. The problem of combustion control has proved to be considerable, and the necessity is seen of always having to carry air mass flow sensing equipment and associated fine metering devices.

Reliability of all the generalized studies of ramjet range made to date is thus brought into question. The refinements in design necessary to solve the foregoing problems and the flight limitations so far experienced make it unlikely that current paper solutions will be fulfilled in all respects and that the time scale for the development of useful weapons from the standpoint of propulsion alone may be many years.

III. DEVELOPMENT STATUS:

The following review of the present state of development of ramjets gives a measure of the difficulties that will have to be surmounted to arrive at a suitable design for long-range ramjets.

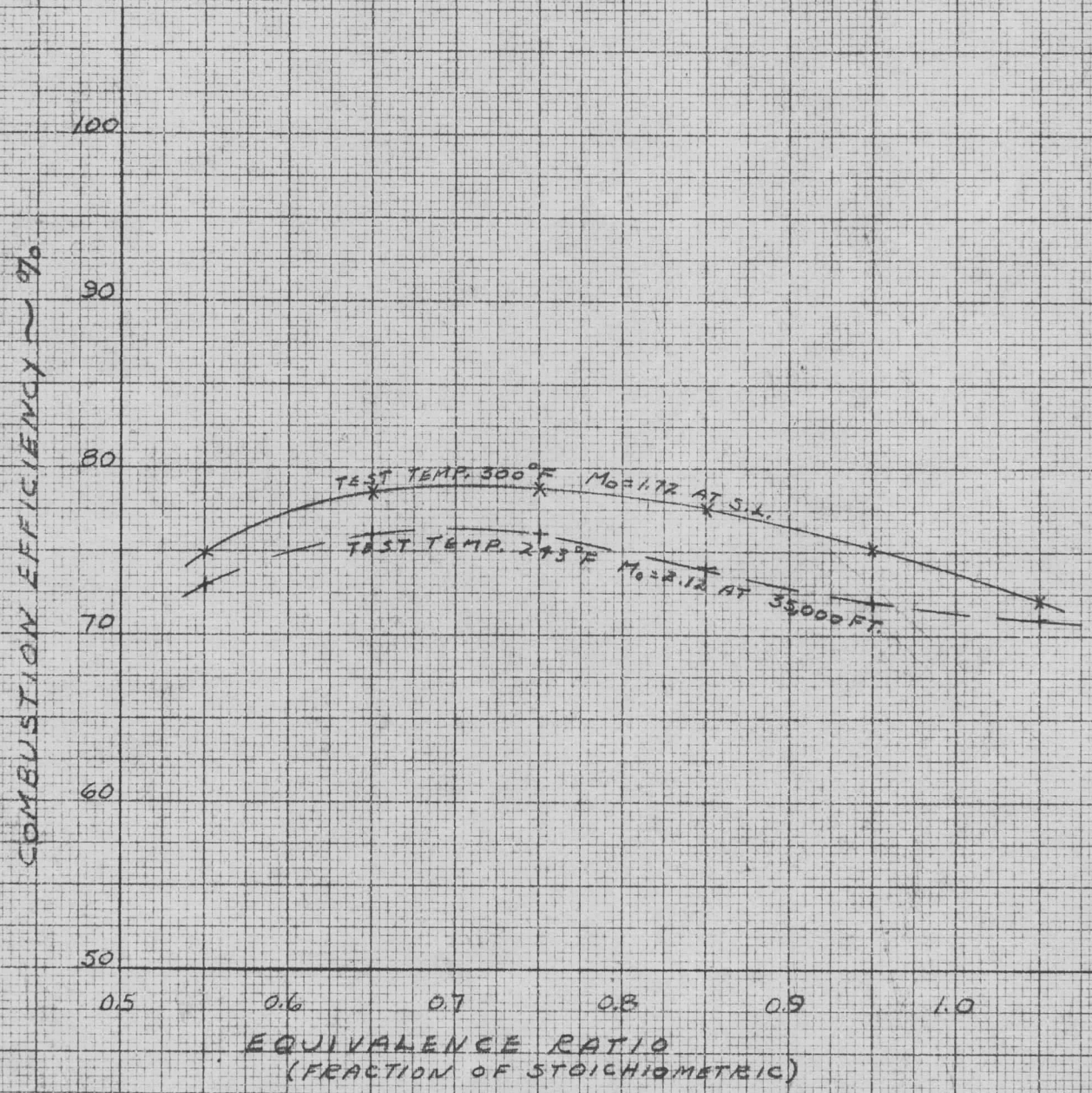
- A. Laboratory Tests: Observations during laboratory tests of propulsion units have lead to criteria of burner performance based upon minimum flight requirements of thrust and maneuverability, and what appear at the present laboratory stage to be practicable minima of burner performance. This laboratory experience in balancing combustion efficiency, air-fuel ratio range, practicability of manufacturing component parts, cost and time of development and general practical limitations has lead to a definition of a satisfactory burner as one which will attain a combustion efficiency greater than 75 percent over an air/fuel range of 2/1, e.g., from $a/f = 30/1$ to $a/f = 15/1$. A typical burner performance curve is presented in Figure 4. It is further required that it burn readily available fuel and meet severe starting conditions. This standard of efficiency and burning range while satisfactory for short-range ground-to-air missiles is by no means adequate for a long range missile where combustion efficiency should be around 90 percent.

This laboratory experience has generally been limited to ramjet engines with a tailpipe exit diameter of twenty-four inches or less at burner pressures greater than one atmosphere, while long-range ramjets will certainly be larger (it is generally estimated that they will be from four to six feet in diameter) and also must operate with burner pressures less than sea level atmosphere.

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Figure 4. Combustion Efficiency vs Equivalence Ratio For a Typical Laboratory Burner. Six - inch scale Designed For $M_0 = 1.6$, Maximum Thrust at Stoichiometric. (Maximum Thrust For This Burner Occurs at Equivalence Ratio 1.2)



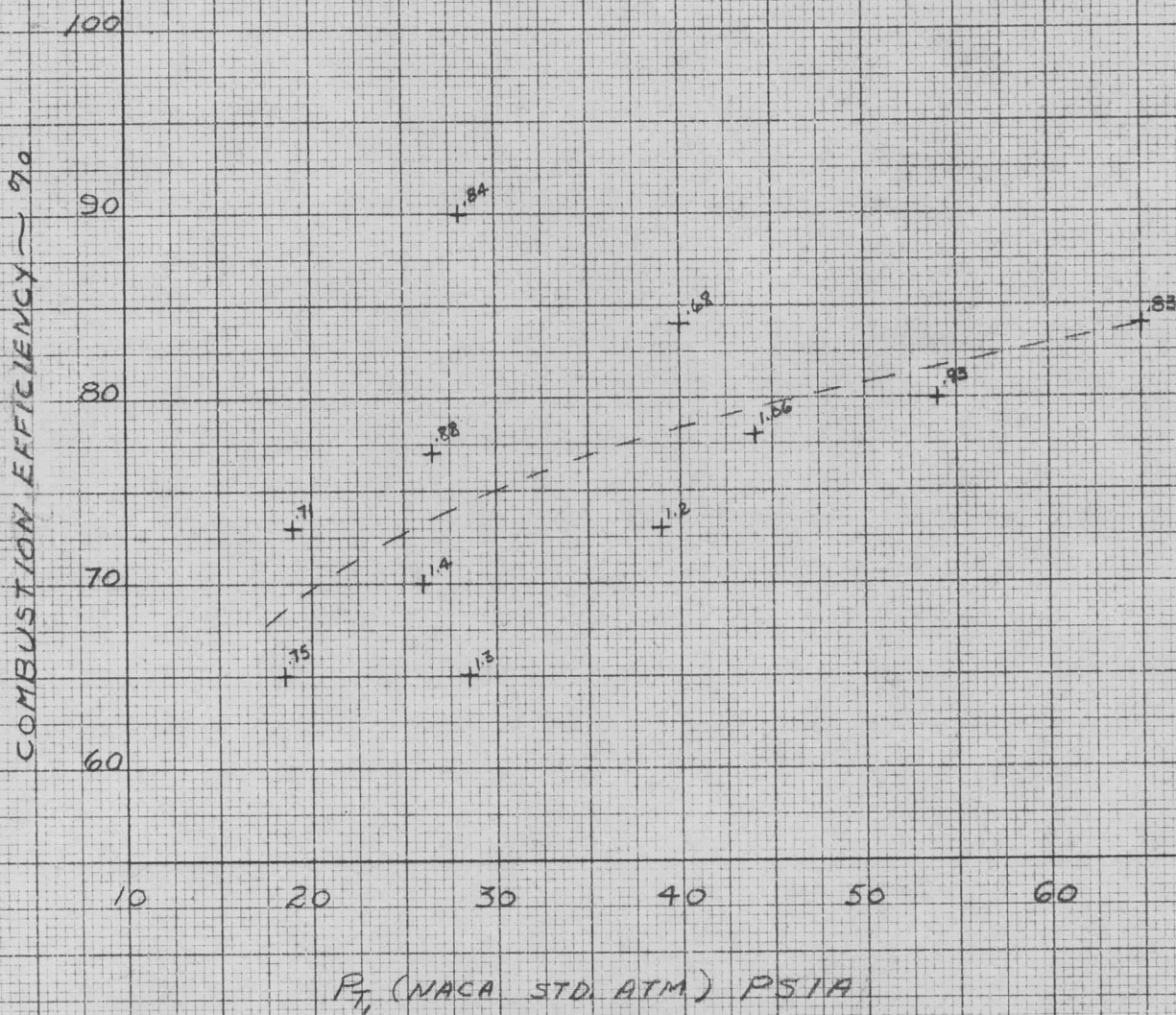
KEUFFEL & ESSER CO.

NO. 359-11. 10 x 10 to the half inch, 5th lines accented.
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Figure 5. Combustion Efficiency vs. Subsonic Diffuser Inlet Stagnation Pressure for Four (4) Flights of Bumblebee BTV, with Pressures Calculated From Normal Shock Rise and Based on NACA Standard Atmosphere at Flight Altitude. (Numbers Indicate Equivalence Ratio)



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- B. Flight Test: Excellent correlation between flight performance and laboratory performance of burners has been found; however, flight has been limited to the region where it is possible to adequately duplicate flight conditions in readily available test facilities. The best example of this limitation has been high altitude flight, where ramjets have stopped burning at from 30,000 to 45,000 feet, which is also the general limitation of readily available test equipment.

Flight test is, of course, far too expensive for burner development tests unless the burner is in a pretty advanced stage already.

- C. Scale of Burner: Experience in scaling burners has clearly indicated that numerous laboratory tests are required of a new size of combustor before satisfactory operation is approached. For example, when a working 18 inch "Bumblebee" burner was scaled up to 24 inches, about twenty weeks of testing over a period of nine months was required before the 24 inch burner reached the standards of performance of the 18 inch burner. This need for testing requires that considerable investment be made in new facilities, with an attendant long and expensive training, research and development period before ramjets any larger than those currently being developed can be built.

- D. Starting Problems: Starting tests in laboratory facilities cannot duplicate all the factors involved, so for this reason a sizeable percentage of loss of valuable flight missiles must be expected in launching failures. Several such failures of 18 inch Bumblebee flights occurred before a proper method of correlating laboratory and flight starting characteristics was found and a solution to the problem devised. The two basic problems of starting are:

1. Ignition under very high combustion velocities, unless a disposable restrictor is used.
2. Delay in developing full thrust, which may mean that the missile decelerates to below a speed from which it can accelerate under its own power.

- E. Altitude Limitations: Although this limitation is mentioned under Section III, B, it represents such a complex of unsolved problems that special emphasis is required. The fact that combustor performance deteriorates rapidly at pressures less than one atmosphere presents a problem which is made doubly hard to solve because the basic factors causing this behavior have not been adequately determined. Figure 5 shows the tendency of combustion efficiency to decline with reduced pressure even at pressures greater than one atmosphere. Research and development of this phase has been severely retarded by lack of test facilities leaving the development of a high altitude propulsion unit several years away.

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F. Control Problems: Fuel flow in a ramjet is, of course, a function of flight velocity and altitude requiring that sensitive air and fuel measuring and metering devices be developed. The development of such controls has presented a considerable problem in the past, and will become even more involved in the future as ramjets have to operate at a wider range of altitudes.

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BOOSTERS FOR LONG RANGE RAMJETS

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BOOSTERS FOR LONG RANGE RAMJETTS

Ramjet efficiencies necessary to make long ranges possible can only be achieved around Mach numbers of the order of 3.

Furthermore these efficiencies require the use of diffusers with poor "off design" characteristics.

A self accelerating ramjet would therefore entail the extreme complication of variable inlets variable nozzles and variable combustors and the elaborate development program leading to their design.

Barring this complication the only alternative is to boost the ramjet to a Mach number close to 3. The pressure in the combustion chamber at such a Mach number is about thirty (30) times the ambient pressure. Therefore launching a ramjet at this high a Mach number also means launching it at altitude lest combustion chamber pressures lead to prohibitive structural weight.

To boost a missile of the size required to both high velocity and high altitude the only means available to date is the liquid propelled, turbo pump rocket. All the design problems involved in the construction of such a booster i.e. power plant, stabilization, structural and aerodynamic problems are the same as those that have to be solved for a pure rocket type missile.

In addition to these the ramjet designer is faced with the assignment of solving the same problems - power plant, structure, stabilization and guidance, aerodynamics - at least equally difficult relating to the ramjet proper.

Of these, contrasting with the rocket the power plant problem involves years of development work and millions of dollars of testing equipment.

The construction of the composite missile will involve the building of two different type power plants, two fuel systems, two stabilization and guidance systems. The probability of malfunctioning of these items is also doubled.

In short, due to the necessity of a high altitude high velocity boost the effort required to produce a long range ramjet is roughly twice that for a rocket missile, both from the standpoint of engineering and the standpoint of construction. Comparing ramjet performance to the performance of a rocket with glide path, it would appear that the benefits derived from this extra effort are questionable at best.

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PROPOSED METHOD FOR IMPROVING
HIGH ALTITUDE PERFORMANCE OF
PRESENT RMI 20,000 POUND THRUST ROCKET

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EFFECT OF NOZZLE EXPANSION ON ROCKET THRUST

In report ZP-48-35003, page 78, mention is made of increasing the missile range by increasing the expansion ratio of the rocket nozzles. It is proposed that this be accomplished without changing the design of the present motor.

The present nozzle, with a 16.63 inch exit diameter, over expands the exhaust up to 5,000 feet and above this altitude expansion is incomplete. By increasing the exit diameter, and allowing more complete expansion at high altitude, rocket thrust can be appreciably increased. However, to prevent severe thrust losses at low altitude, an annular plug must be inserted in the exit nozzle so that the excess area is filled with a solid resistant material (probably graphite) to prevent over expansion. At 20,000 feet, the plug is released and the thrust from the greater expansion is provided.

Figure 6 shows the thrust increase with altitude for the present nozzle design and the gain that can be expected if greater expansion is used.

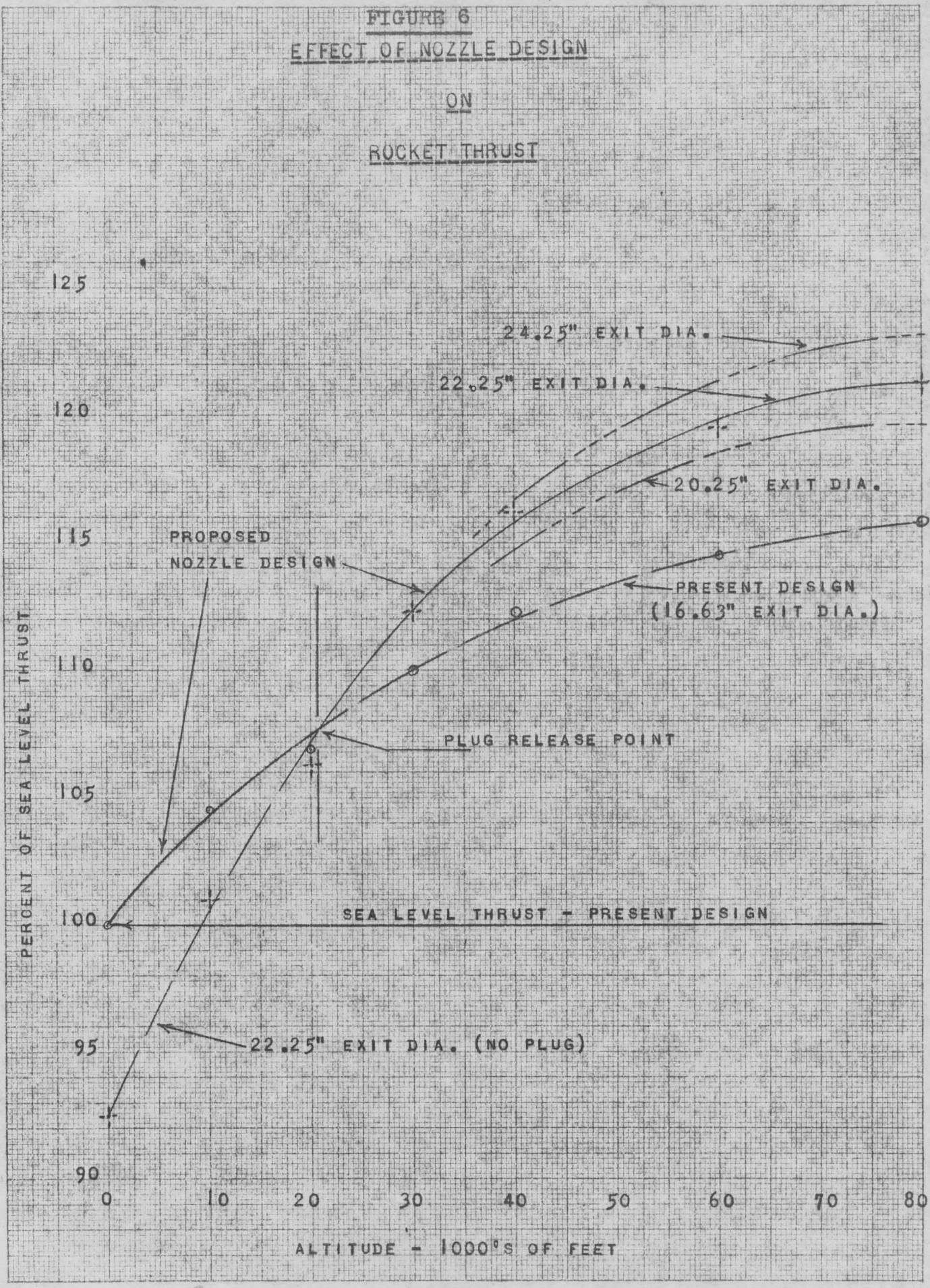
Figure 7 is a schematic drawing of the proposed nozzle extension and plug insert as related to the present rocket design. The only change is an addition to the basic design.

The following table shows the percent gain in thrust by expanding beyond the present exit diameter. The Figures represent the net gain from sea level to 520,000 feet altitude (using a plug during the first 20,000 feet).

TABLE

<u>DESIGN</u>	<u>EXIT DIAMETER (INCHES)</u>	<u>PERCENT THRUST GAIN</u>
#1 (present)	16.63	0
#2	20.25	3.1
#3	22.25	4.5
#4	24.25	6.0

FIGURE 6
EFFECT OF NOZZLE DESIGN
ON
ROCKET THRUST



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Millimeters, 6 mm. lines argenteo, cm. lines heavy.
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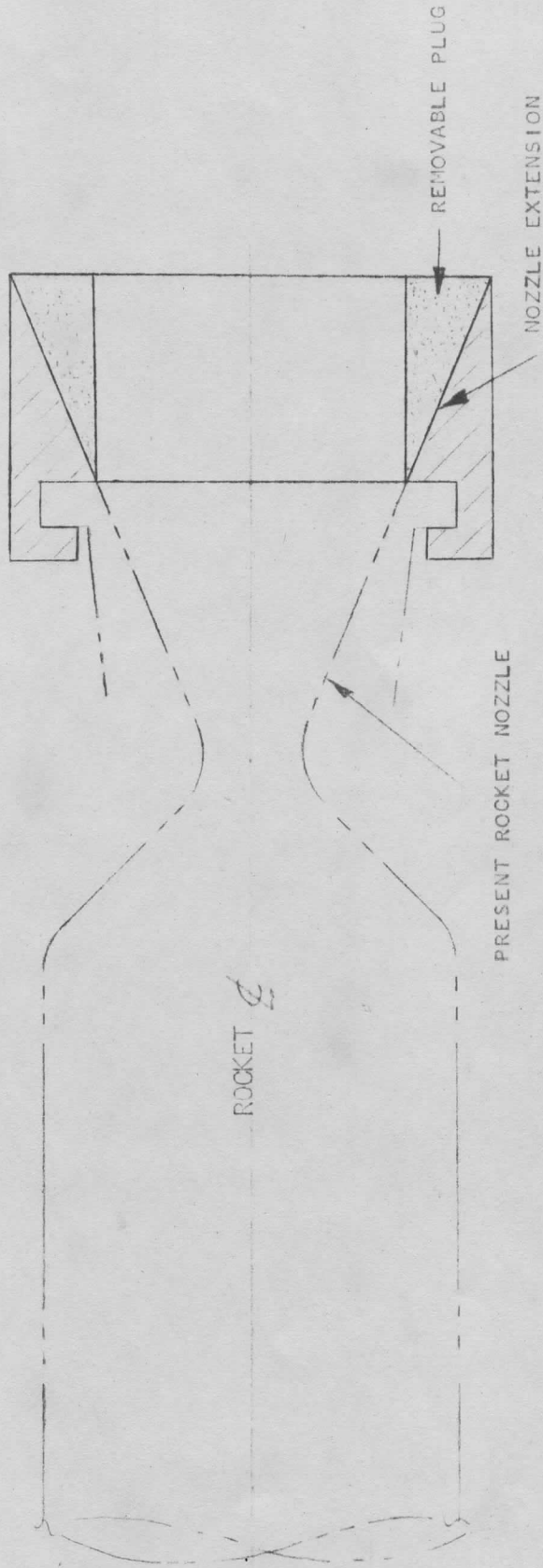


FIG. 7 SCHEMATIC ——— PROPOSED ROCKET NOZZLE EXTENSION AND REMOVABLE PLUG

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DISCUSSION:

Rocket operation is to be maintained from sea level to 520,000 feet altitude. Information on pressures at high altitude was obtained from NACA report number 1428 and other references i.e., Rand Project, Whipple, Betz Et Al, and Warfield. Pressure ratios were then determined by using chamber pressure at 300 psia and various altitude pressures. Rocket thrust was calculated by the formula:

$$F = C_F P_C a_T \text{ (ref. *)}$$

where: F = thrust of rocket

C_F = thrust coefficient

P_C = chamber pressure

a_T = nozzle throat area

In each design shown in the above table, the throat area is the same (47.1 sq. in.). In Figure 6 the thrust curves were terminated at 80,000 feet, since above this altitude, the thrust coefficient is constant.

The larger exit diameters give higher thrust values at high altitude at the expense of a low altitude loss. However the use of an annular plug (see Figure 7) appears to be a satisfactory solution. The plug is designed to maintain the 16.63 inch diameter of the present nozzle exit up to 20,000 feet. At this altitude it will be released, leaving the expanded nozzle which will give the higher thrust values above this point.

Design number 3 is approximately 6 percent below optimum at 80,000 feet, however a greater nozzle diameter will cause an external (aerodynamic) drag that will probably more than offset the thrust gain. Above 80,000 feet the thrust of a given design will remain constant while optimum thrust increases. Values of optimum thrust above this altitude cannot be found with the available data.

Figure 6 shows thrust values for three possible extensions to the present nozzle. The solid curve labelled "proposed nozzle design" shows the probable best nozzle design for the 1,000 mile missile in its present configuration.

The nozzle extension will not require a redesign of the present rocket and all its heat balance problems. The extension can be made of graphite (or some other heat resistant material) and attached to the present nozzle lip.

Ref. California Institute of Technology Reference Text Book "JET PROPULSION" Eq. 5.17.

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WARHEAD SKIN TEMPERATURES

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WARHEAD SKIN TEMPERATURES

Figure 8 shows the temperatures anticipated for the warhead skin for the ballistic and glide path flights. It can be seen from the curves that the maximum temperature in the ballistic warhead occurs shortly before impact. In the glide type warhead the maximum occurs during the first quarter of the glide range. The skin temperature then decreases during the remainder of the flight. The conditions of the glide path are somewhat less severe, lower skin temperatures and a longer heating period alleviating thermal stresses in the skin considerably.

Materials and techniques currently in use are suitable for the design of a warhead skin for a 1000 mile range ballistic path missile, or a 2000 mile range glide path missile.

The method used to determine the data of Figure 8 is essentially that of Convair Report No. DEVF 346-17. This is a step-by-step method including the effects of heating by the boundary layer, dissociation in the boundary layer, radiation by the skin, and the thermal capacity of the skin.

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WARHEAD SKIN TEMPERATURES AS A FUNCTION OF FLIGHT PLAN

- Ⓐ BALLISTIC TRAJECTORY ~ .063" THICK STAINLESS STEEL SKIN
- Ⓑ GLIDE PATH FLIGHT ~ .051" THICK STAINLESS STEEL SKIN

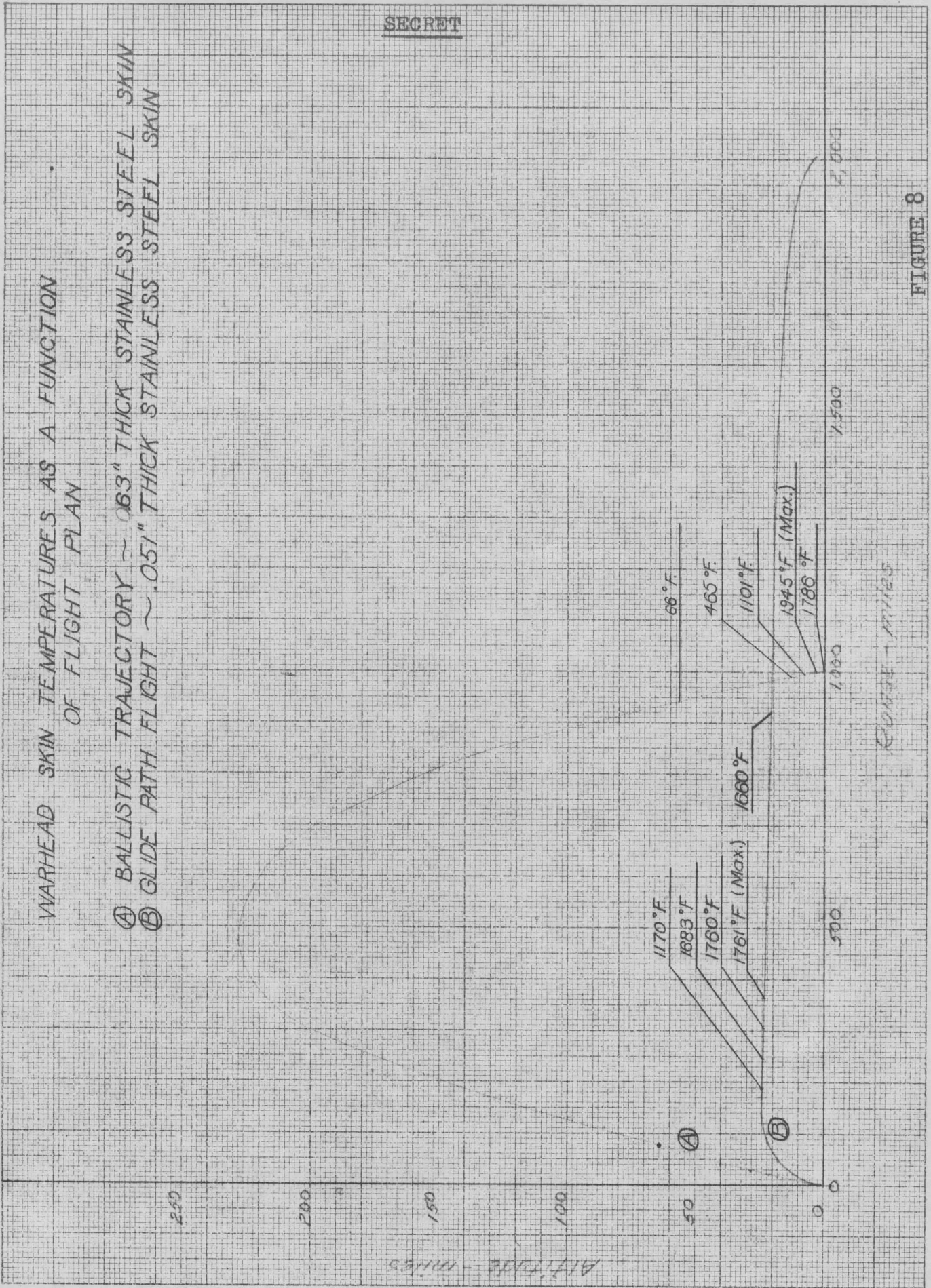


FIGURE 8

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 Millersburg, Ohio
 No. 100-48

DATA SHEETS

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ROCKET MISSILE OF 2000 MILE RANGE

WITH

6000 POUND GLIDE TYPE WARHEAD

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A. General: The 6000 pound warhead 1000 mile ballistic missile described in CVAC Report No. ZP-48-35003 may be considered a first step toward the ultimate development of a glide type warhead missile of roughly double the range.

This possibility for future development was lightly touched upon in Report ZP-48-35003 on pages 2 and 31. Some results of further preliminary study on this problem can now be presented.

Time has not permitted an independent computation of skin friction drags at high Mach numbers during the glide. These computations are very elaborate. Therefore the performance analysis shown is based on the use of skin friction coefficients as determined by RAND.

The establishment of an optimum configuration for the glide type missile was found to be considerably more difficult than for the ballistic type. The warhead size and weight, maximum L/D, altitude and velocity at burnout, warhead skin temperatures are all interdependent parameters. By a process of compromise, a first approximation for an optimum configuration, shown on pages 29 and 30 was arrived at. This configuration can doubtlessly be improved by further study.

B. Description:

Overall Length	83.5 ft
Body Diameter	6.75 ft
Gross Weight	74775 lb
Warhead	6000 lb
Fabricated Wt	6635 lb
Fuel Wt	62140 lb
Take off Thrust	108,000 lb
Range (Approx.)	2000 St Miles

The power plant is the same as that of the proposed ballistic missile including the method of jettisoning the booster stage.

Inasmuch as guidance is necessary during the glide, all intelligence equipment is located in the warhead.

Aluminum construction is used for the alcohol tank section, aft body and empennage. The latter is jettisoned at the end of the booster stage.

The gliding warhead structure separates at burnout and is built out of stainless steel due to temperature considerations. The aft part of this structure is the oxygen tank. The shape of the warhead body is aerodynamically stable at high Mach numbers so that no tail surfaces are required during the glide. Roll stabilization is provided by a pair of rollerons located at the base of the nose cone. During the glide trim in pitch and yaw is controlled by a seiveling (canard type) nose.

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C. Weight Statement:

GROSS WEIGHT SUMMARY

6000 LB GLIDE TYPE WARHEAD

	<u>STAGE I</u>			<u>STAGE II</u>		
	<u>WEIGHT</u>	<u>ARM</u>	<u>MOMENT</u>	<u>WEIGHT</u>	<u>ARM</u>	<u>MOMENT</u>
<u>Gross Weight</u>	74775	626	46864560	17492	555	9709568
<u>Weight Empty</u>	(12635)	(515)	(6499560)	(9926)	(396)	(3926560)
Payload	6000	265	1590000	6000	265	1590000
Body aft of separation	799	860	687400	554	820	453500
Fins	100	989	98900	---	---	---
Power plant	3197	946	3027360	1039	945	981360
Equipment Nose	715	215	154000	715	215	154000
Equipment tail	349	910	317500	143	863	123300
Body fwd of separation	1475	423	624400	1475	423	624400
<u>Fuel</u>	(62140)	(650)	(40365000)	(7566)	(764)	(5783000)
H ₂ O ₂	1243	936	1165000	151	949	143000
Lox	33832	507	17200000	4119	670	2760000
Alcohol	27065	811	22000000	3296	876	2880000
	60897					
Wt. at end of 1st stage	20201	610	12282560			
Wt. at end of 2nd stage				9926	396	3926560
Warhead weight				8190	290	2368400

20201
 17962
 2259

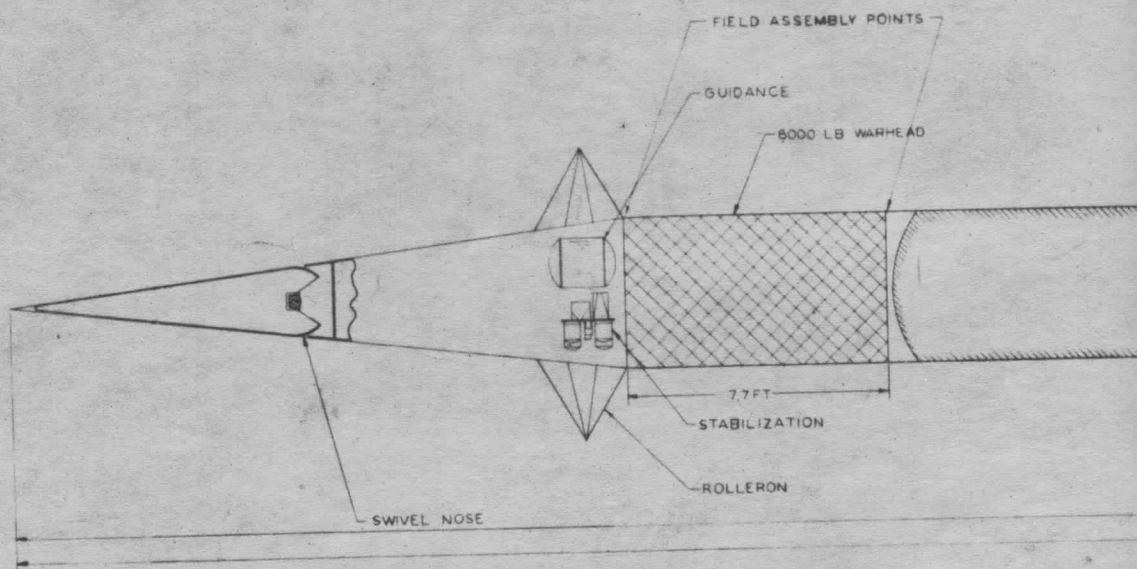
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WEIGHT EMPTY
WEIGHT AND BALANCE
6000 lb Glide Type Warhead

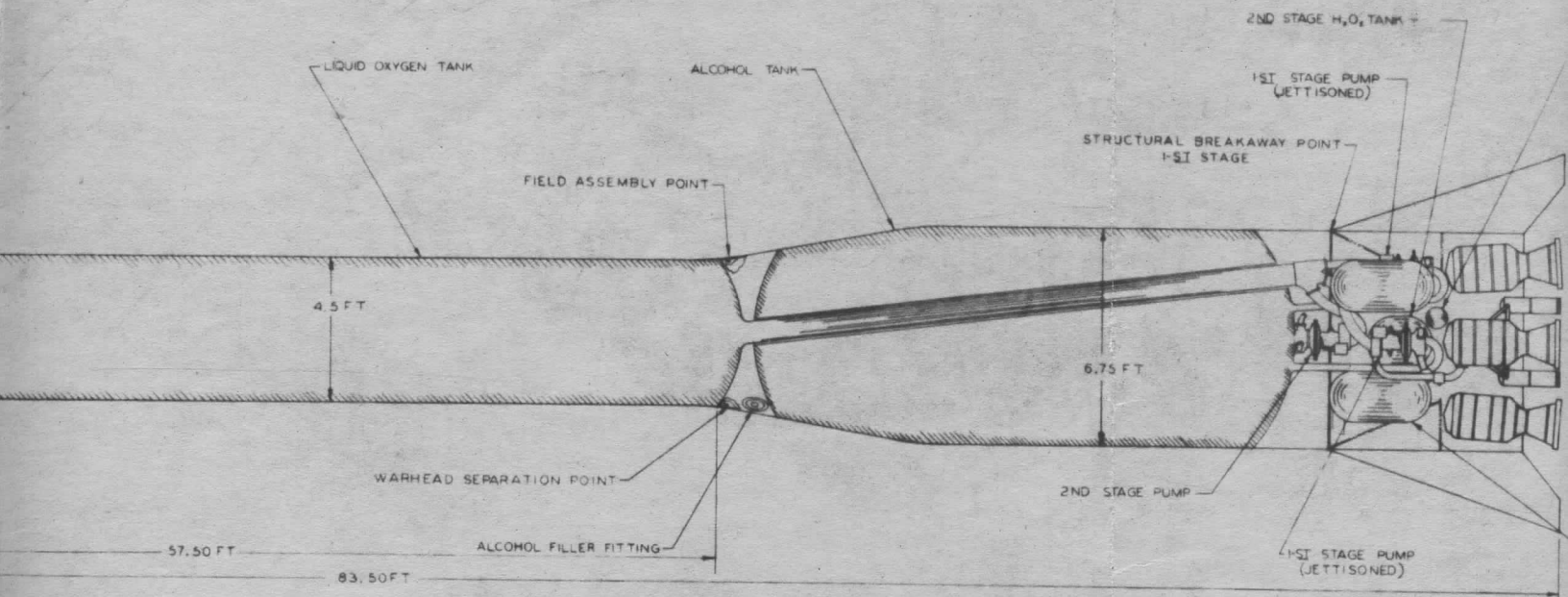
ITEM	STAGE I			STAGE II		
	WEIGHT	ARM	MOMENT	WEIGHT	ARM	MOMENT
<u>Glide Warhead</u>	(8190)	(290)	(2368400)	(8190)	(290)	(2368400)
Nose Cone	269	146	95000	269	146	95000
Warhead	6000	265	1590000	6000	265	1590000
Fixed EQ + Guid	545	182	99000	545	182	99000
Trimming Dev.	100	109	10900	100	109	10900
Skin	1138	435	495000	1138	435	495000
Residual Fuel	50	680	34000	50	680	34000
Fittings	20	507	10100	20	507	10100
Tank Ends	68	507	34400	68	507	34400
<u>Power Plant</u>	(3197)	(946)	(3027360)	(1039)	(945)	(981360)
<u>Motor & Prop Valves</u>	(1225)	(974)	(1192600)	(457)	(975)	(445600)
(4) 20000# Thrust	768	973	747000	---	---	---
(1) 20000# Thrust	192	973	187000	192	973	187000
(4) 2000 # Thrust	165	988	163000	165	988	163000
(4) 20000# Thrust Prop Valve	80	956	76500	80	956	76500
(1) 20000# Thrust Prop Valve	20	956	19100	20	956	19100
<u>Pumps</u>	(725)	(935)	(679000)	(145)	(916)	(133000)
(4) 20000 Thrust	580	941	546000	---	---	---
(1) 28000 Thrust	145	916	133000	145	916	133000
<u>Mounts</u>	(210)	(935)	(196600)	(110)	(931)	(102500)
80000 Mount	100	941	94100	---	---	---
28000 Mount	110	931	102500	110	931	102500
<u>Valves and Regulators</u>	(160)	(941)	(150600)	(40)	(941)	(37600)
28000	40	941	37600	40	941	37600
80000	120	941	113000	---	---	---

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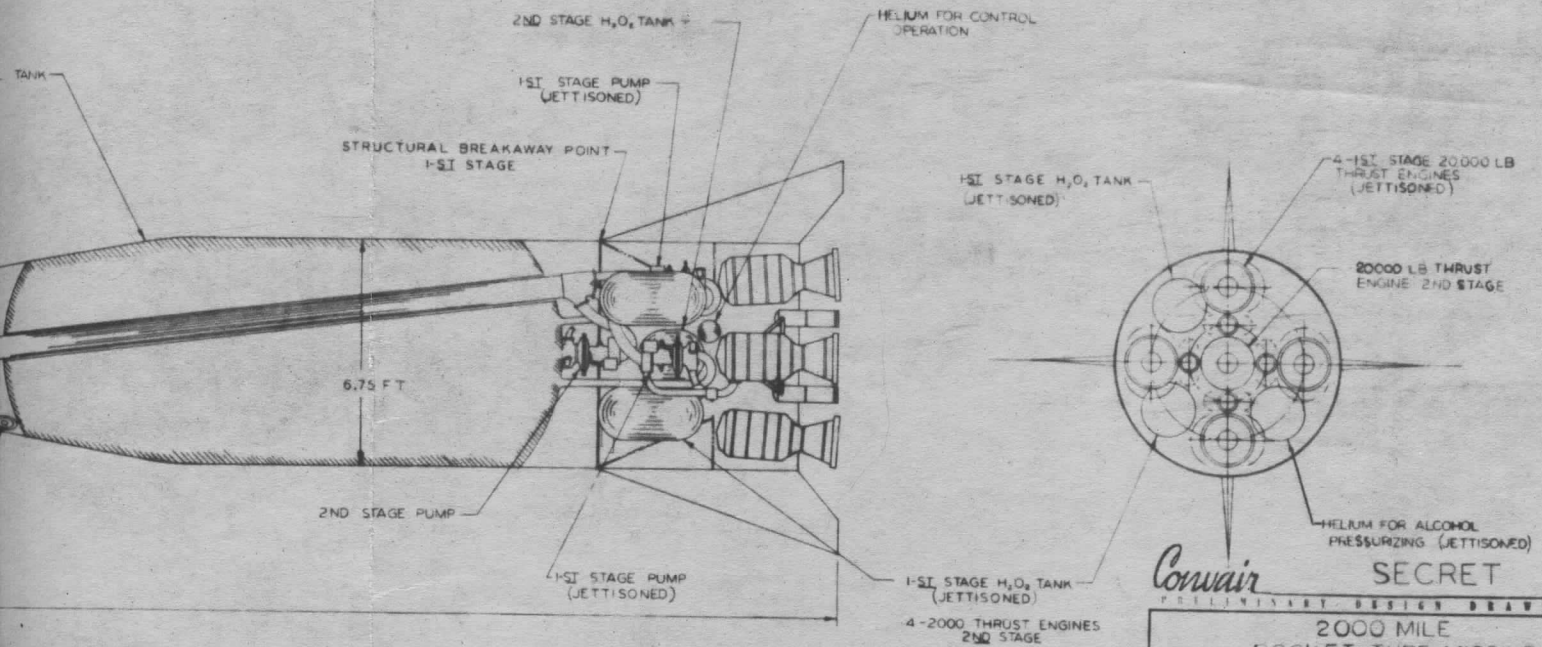
	STAGE I			STAGE II		
	WEIGHT	ARM	MOMENT	WEIGHT	ARM	MOMENT
Lines and Fittings	(365)	(926)	(338000)	(135)	(919)	(124200)
28000	95	925	88000	95	925	88000
80000	270	925	250000	40	905	36200
Press Supply	(150)	(932)	(139860)	(35)	(924)	(323360)
Heat Exchanger - Oxygen	12	925	11100	12	925	11100
Helium Tank Alcohol Pressure	115	936	107500	---	---	---
Helium Tank Control Operation	13	916	11900	13	916	11900
Helium	10	936	9360	10	936	9360
Pump Prop. System	(132)	(937)	(123700)	(32)	(941)	(30100)
(2) H2O2 Tank - Large	100	936	93600	---	---	---
(1) H2O2 Tank - Small	32	941	30100	32	941	30100
Residual Fuel	(230)	(902)	(207000)	(85)	(895)	(76000)
28000	60	902	54000	60	902	54000
80000	170	902	153000	25	880	22000
Fixed Equipment	(349)	(910)	(317500)	(143)	(863)	(123300)
(6) Valves	36	906	32600	18	906	16300
(6) Response Units	30	906	27200	15	906	13600
8000 Swivel Cont.	40	976	39000	40	976	39000
80000 Swivel Cont.	153	946	145000	---	---	---
Staging Controls	40	966	38600	20	966	19300
Warhead Eject.	50	702	35100	50	702	35100
Aft Body	(799)	(860)	(687400)	(554)	(818)	(453500)
A/c Tank	392	811	318000	392	811	318000
Aft Bulkhead	39	902	35200	39	902	35200
Fwd Bulkhead	23	710	16200	23	710	16200
Lox. Tunnel	60	800	48000	60	800	48000
Engine Struct.	285	946	270000	40	906	36100
Fin	100	989	98900	---	---	---



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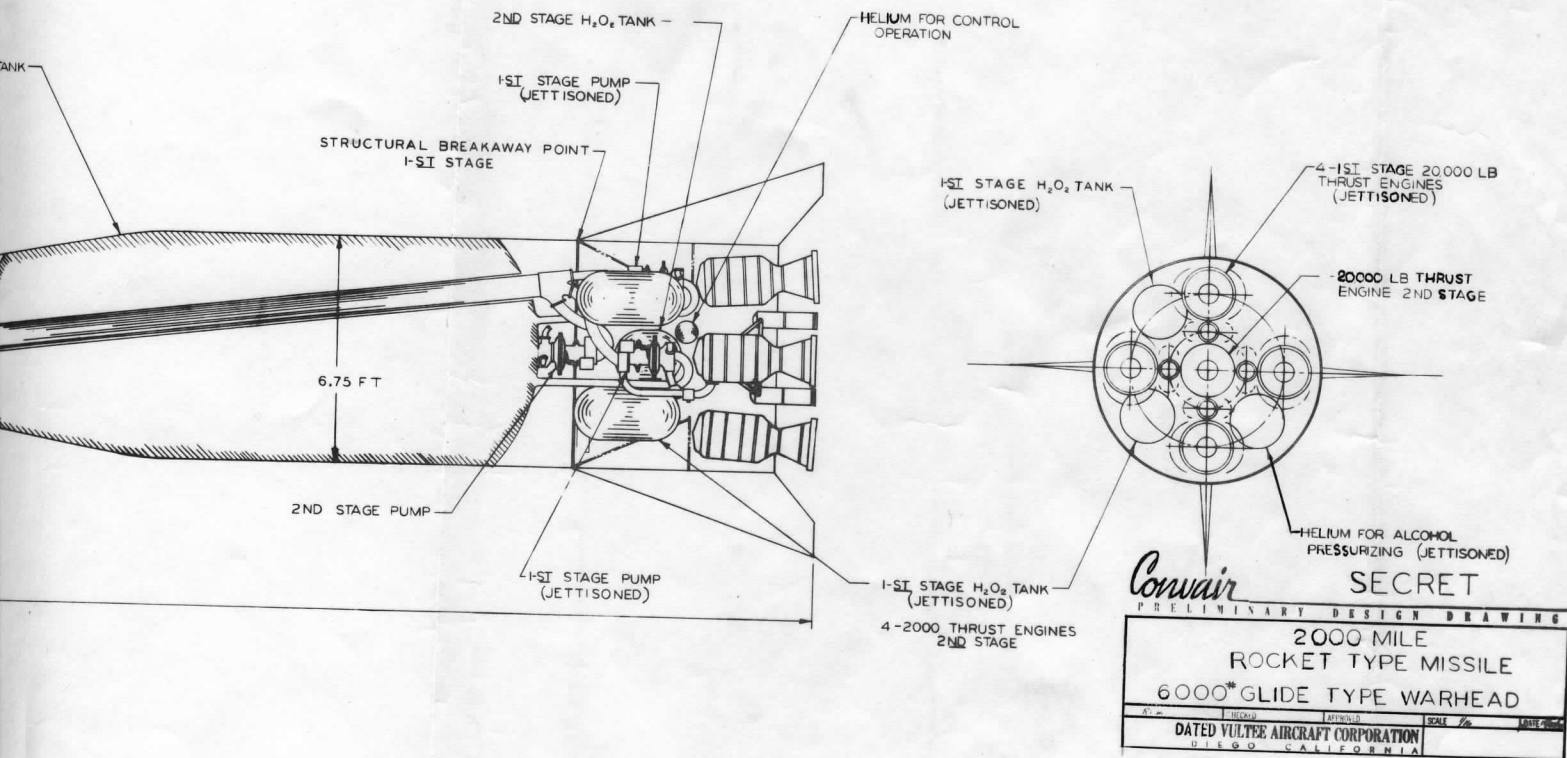
SECRET



Convair **SECRET**
PRELIMINARY DESIGN DRAWING
2000 MILE
ROCKET TYPE MISSILE
6000⁺ GLIDE TYPE WARHEAD

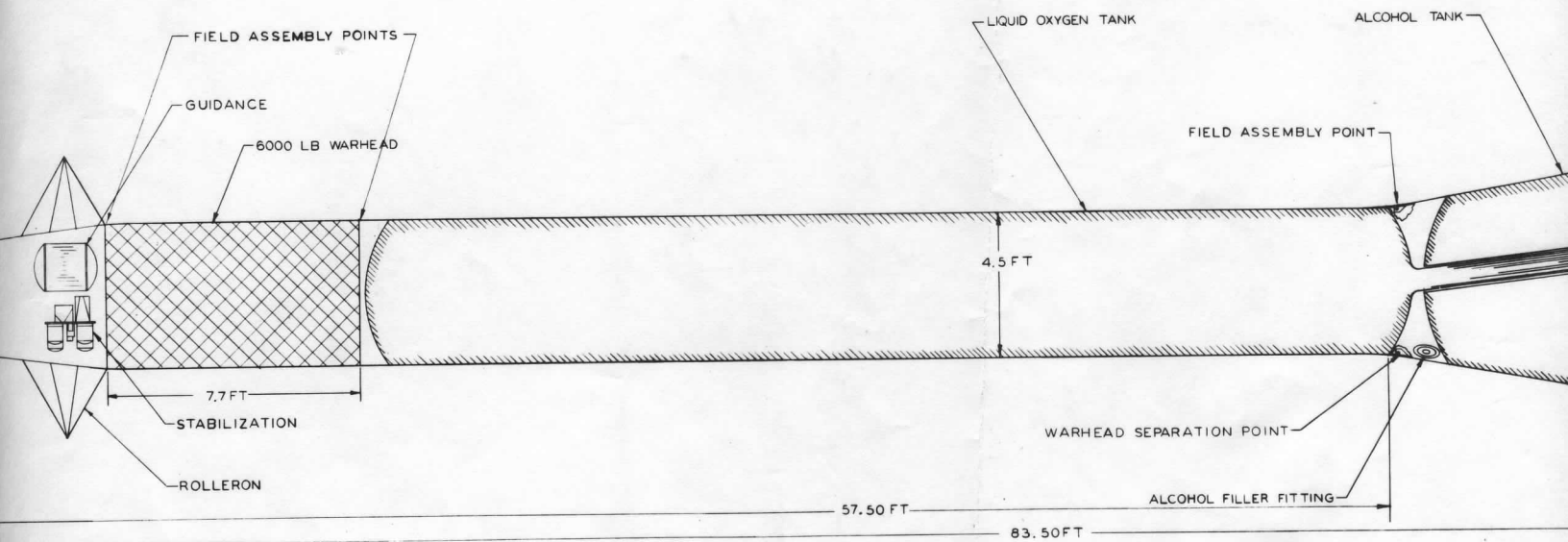
DATE	DESIGN	SCALE	APPROVED

DATED VULTEE AIRCRAFT CORPORATION
MILV - CALIFORNIA

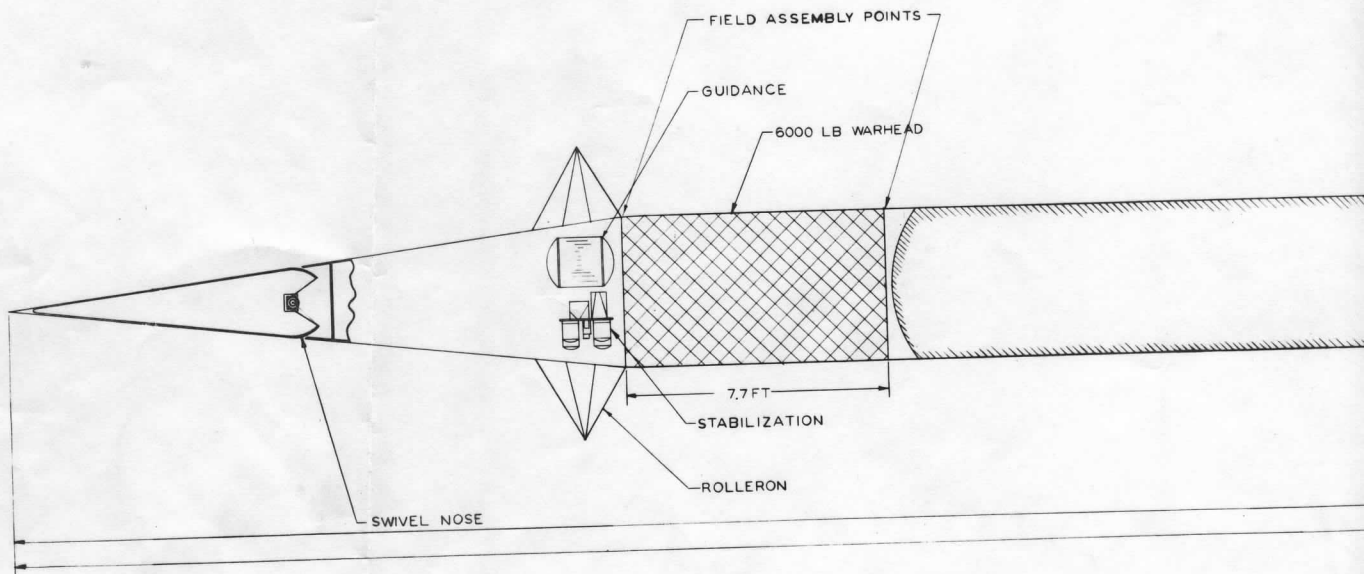


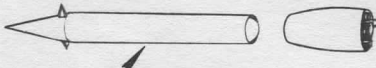
Convair
SECRET
PRELIMINARY DESIGN DRAWING
2000 MILE
ROCKET TYPE MISSILE
6000* GLIDE TYPE WARHEAD
DATED VULTEE AIRCRAFT CORPORATION
HILSD, CALIFORNIA

SECRET

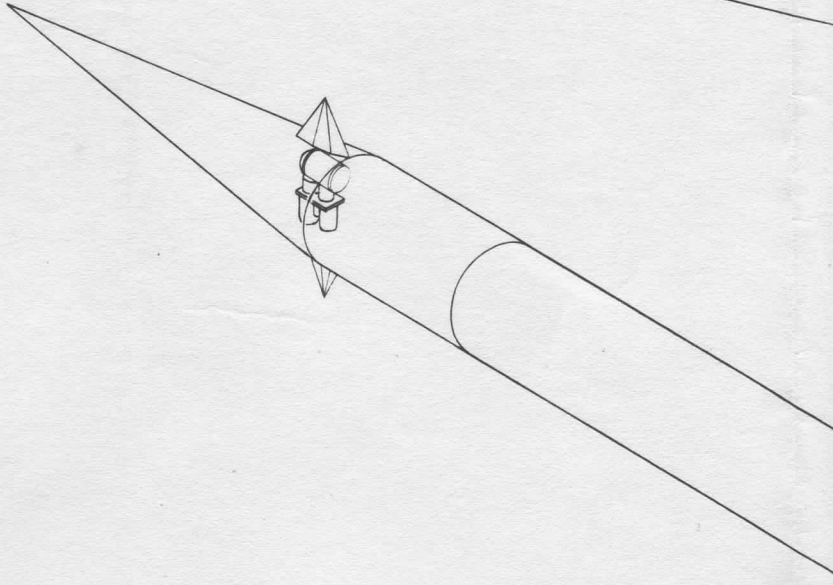


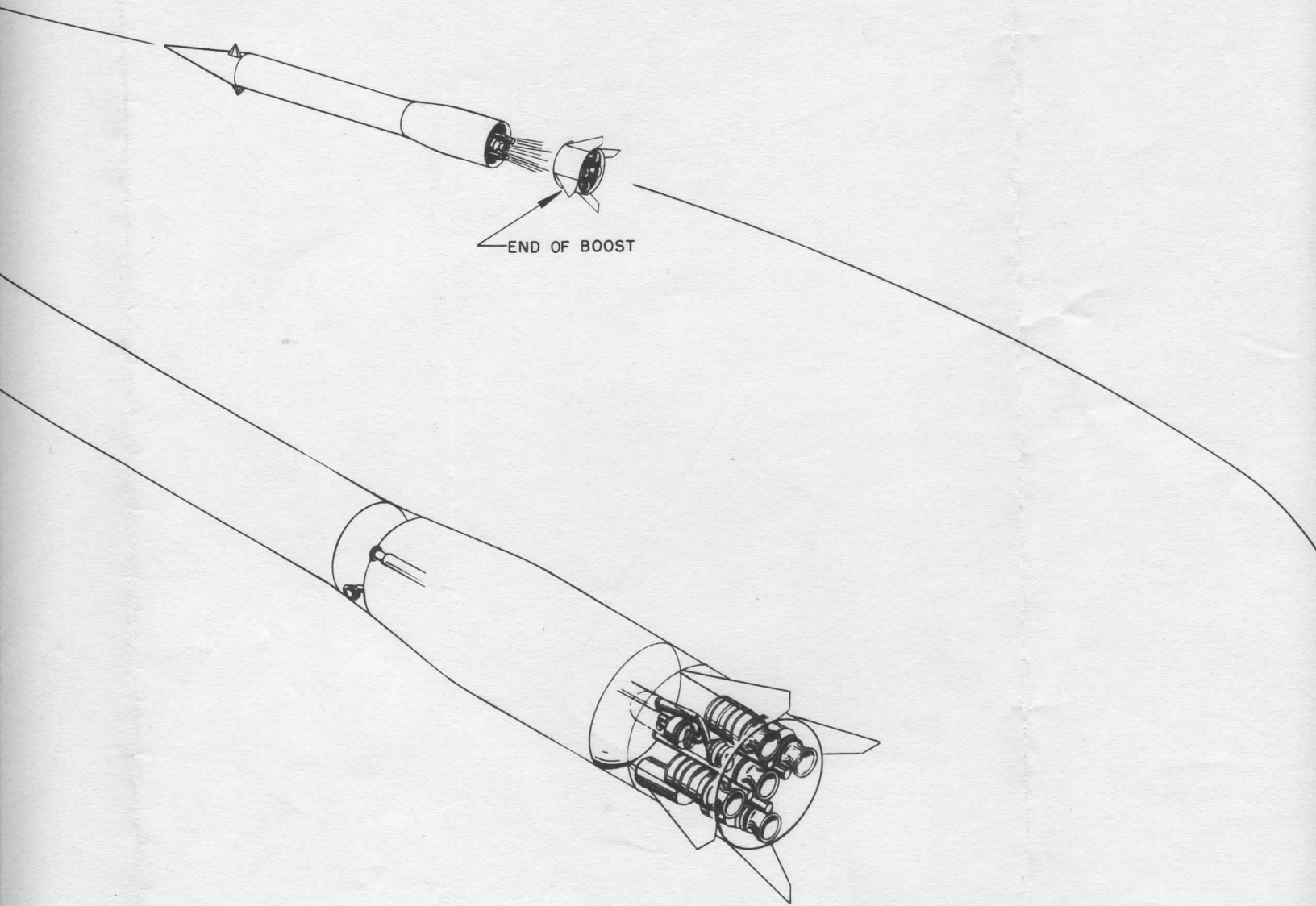
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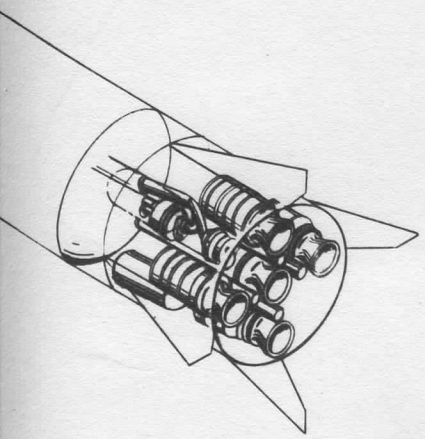
START OF GLIDE







D OF BOOST



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D. Stability and Control: The glide type rocket missile is stabilized by four (4) stabilizing surfaces (fins) during the booster stage portion of the flight. Only very small fins (12 square feet per panel) are required to stabilize the missile during the booster stage and no fins are required during the final stage of burning. This is a result of the excellent stabilizing characteristics of the body along and the forward position of the center of gravity due to the location of the warhead near the nose.

During powered flight the missile is controlled by the use of swiveling rockets in the same manner as noted on page 12 of reference 1.

The glide configuration will be controlled in pitch and yaw by a swiveling (canard type) nose. This will maintain the necessary angle of attack for maximum L/D and keep the missile on the correct azimuth. This appears to be the best type of control in pitch and yaw, however, additional research will be required on this type of control to ascertain the effectiveness of the nose swiveling and its effect on aerodynamic characteristics of the rest of the body at supersonic speeds. It is possible that movable surfaces (fins) may be required to maintain the glide configuration at an angle of attack.

In order to control the glide configuration in roll, a set of roll flippers were placed just forward of the warhead (see page 29). Since no large disturbances in roll are anticipated the resultant roll flippers are correspondingly small.

Lift and Center of Pressure of Powered Missile Configuration:

Body Alone: The lift and moment characteristics for the body alone were determined by the method outlined in reference 2 for the low Mach numbers ($M = 1.25$ to $M = 5.0$) and by use of Newtonian flow characteristics reference 3) for $M > 10.0$. The results obtained using the method shown in reference 2 have shown good agreement with German wind tunnel pressure distribution test on the V-2 and static stability wind tunnel tests on the MX-774 at Aberdeen, at low angles of attack ($\alpha < 10^\circ$). This theory however, breaks down in the vicinity of $M = 5.0$ for normal nose angles. The next logical step then appears to be Newtonian flow characteristics (reference 3) which appear to check the exact gas dynamic theory (reference 3) in the region of $M = 10.0$ to 20.0 .

Since it is believed that the above two theories hold appreciably well for their respective Mach numbers regions and since it is further believed that no discontinuity will occur in the intermediate region (i.e., $M = 5.0$ to $M = 10.0$), the two curves were faired into each other. A plot of normal force and center of pressure characteristics are presented on page 33 for the complete powered configuration.

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Fin Alone: The lift and moment characteristics of the fins were obtained from the method outlined in reference 2 for all Mach number regions in which fins were used (M 7). These results are presented in page .

Complete Configuration: Given the body alone and fin alone lift and moment characteristics above the center of pressure of the complete powered configurations are calculated by

$$\frac{x}{\lambda} = \frac{\frac{dCN_B}{d\alpha}(1 - \frac{XB}{\lambda}) + 2 \frac{dCN_F}{d\alpha} \frac{XF}{\lambda}}{\frac{dCN_B}{d\alpha} + 2 \frac{dCN_F}{d\alpha}}$$

The above equation is used only for the booster stage (Stage I) where fins are necessary. The final stage (Stage II) is stable without fins and thus only body alone characteristics were used.

A plot of the center of pressure and center of gravity versus Mach number of the complete powered configuration for Stages I and II is presented on page. .

Lift and Center of Pressure of Glide Configuration: The glide configuration has a conical nose (half nose angle = 5.0°) and a cylindrical afterbody (see page 29) of 8.7 calibers.

The lift and moment characteristics were obtained by the same method previously outlined for the body alone of the powered flight configuration. However, the results were obtained from higher angle of attack considerations for the glide configuration due primarily to the angles of attack required to maintain maximum Lift/Drag in the glide path.

A plot of $\frac{dCN_G}{d\alpha}$ and $\frac{X_G}{\lambda}$ versus Mach number for the glide configuration is presented on page 35.

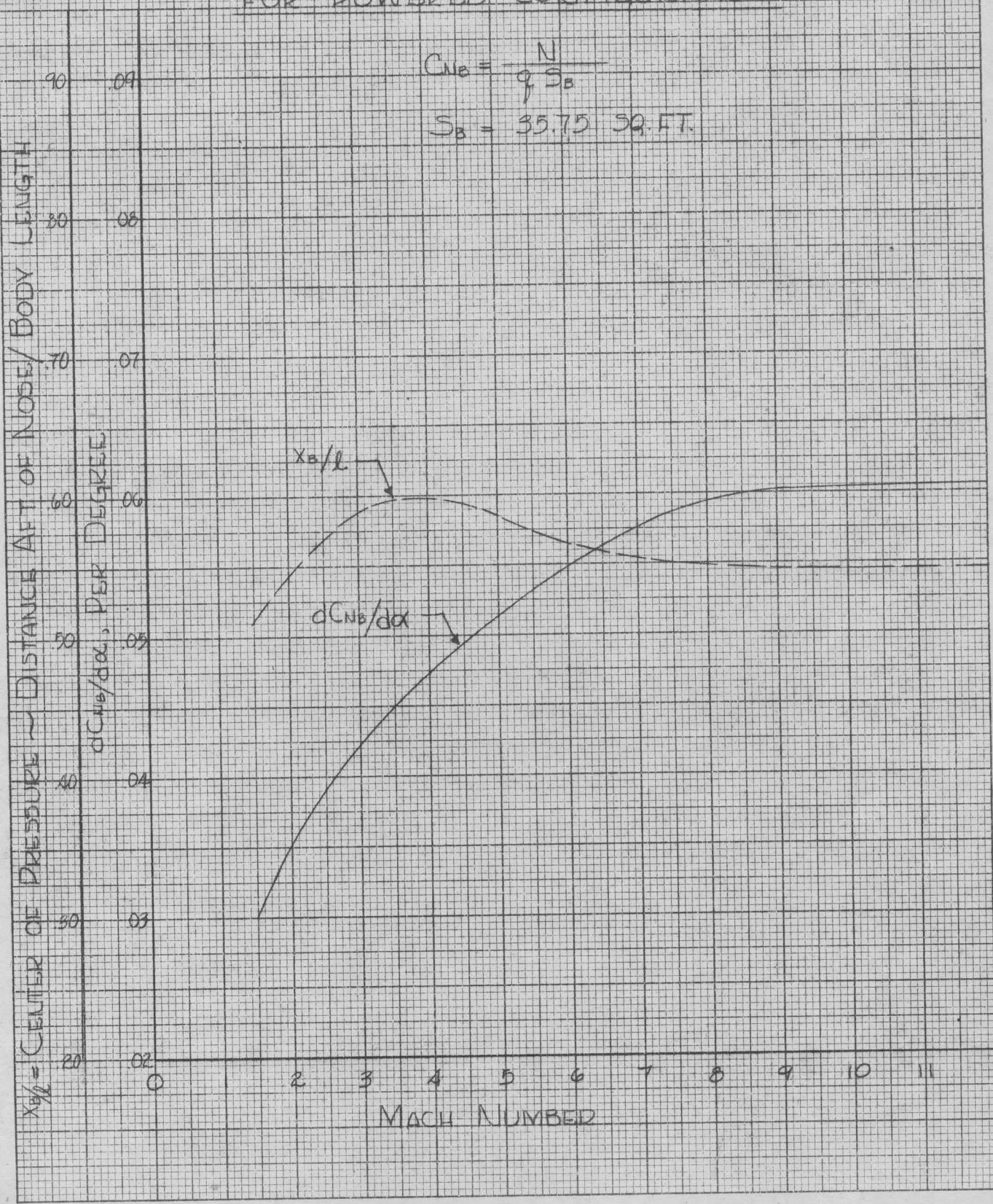
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ROCKET GLIDE TYPE MISSILE
 (6000 LB. WARHEAD ~ 2000 MI.)

VARIATION OF x_{cp}/l & $dC_{Nb}/d\alpha$ WITH MACH NUMBER
FOR POWERED CONFIGURATION

$$C_{Nb} = \frac{N}{q S_B}$$

$$S_B = 35.75 \text{ SQ. FT.}$$



NO. 329-11. 10 X 10 to the first inch, 2 1/2 times extended.
 MADE IN U.S.A.
 Engineering, 1 X 10 in.

KENNEL & ESSER CO.

SECRET

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ROCKET GLIDE TYPE MISSILE

(6000 LB. WARHEAD - 2000 MI.)

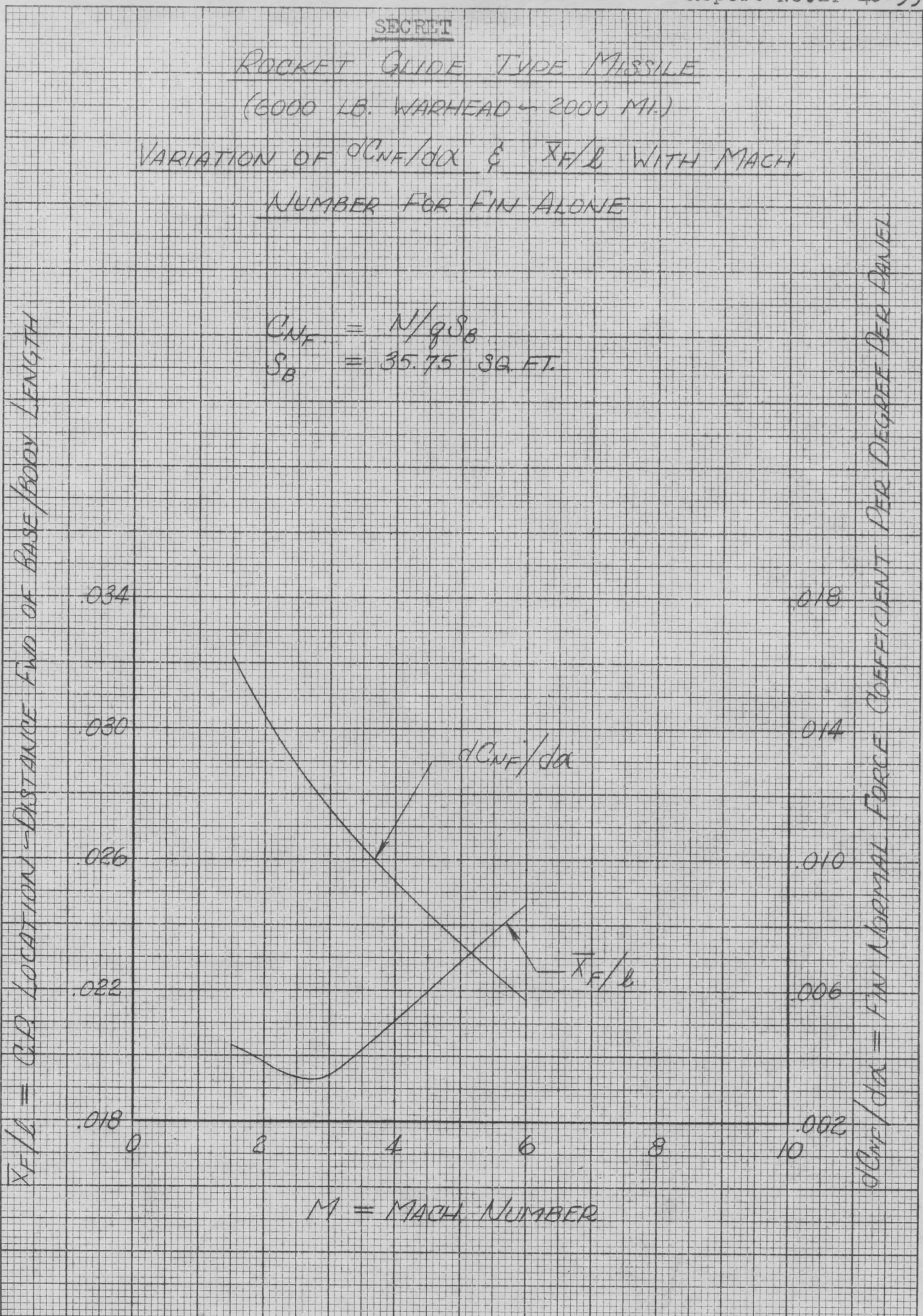
VARIATION OF $dC_{NF}/d\alpha$ & $\bar{x}_{F/L}$ WITH MACH NUMBER FOR FIN ALONE

$$C_{NF} = N/qS_B$$

$$S_B = 35.75 \text{ SQ. FT.}$$

$\bar{x}_{F/L}$ = C.P. LOCATION - DISTANCE FWD OF BASE / BODY LENGTH

$dC_{NF}/d\alpha$ = FIN NORMAL FORCE COEFFICIENT PER DEGREE PER PANEL



M = MACH NUMBER

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NO. 38-11, 10 x 10 to the first inch, 2 1/2 inch lines secured.

KENFLET & ESSER CO.

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ROCKET GLIDE TYPE MISSILE
 (6000 LB. WARHEAD - 2000 MI.)

VARIATION OF x_g/l & $dC_{Ng}/d\alpha$ WITH MACH NUMBER ALONG GLIDE PATH

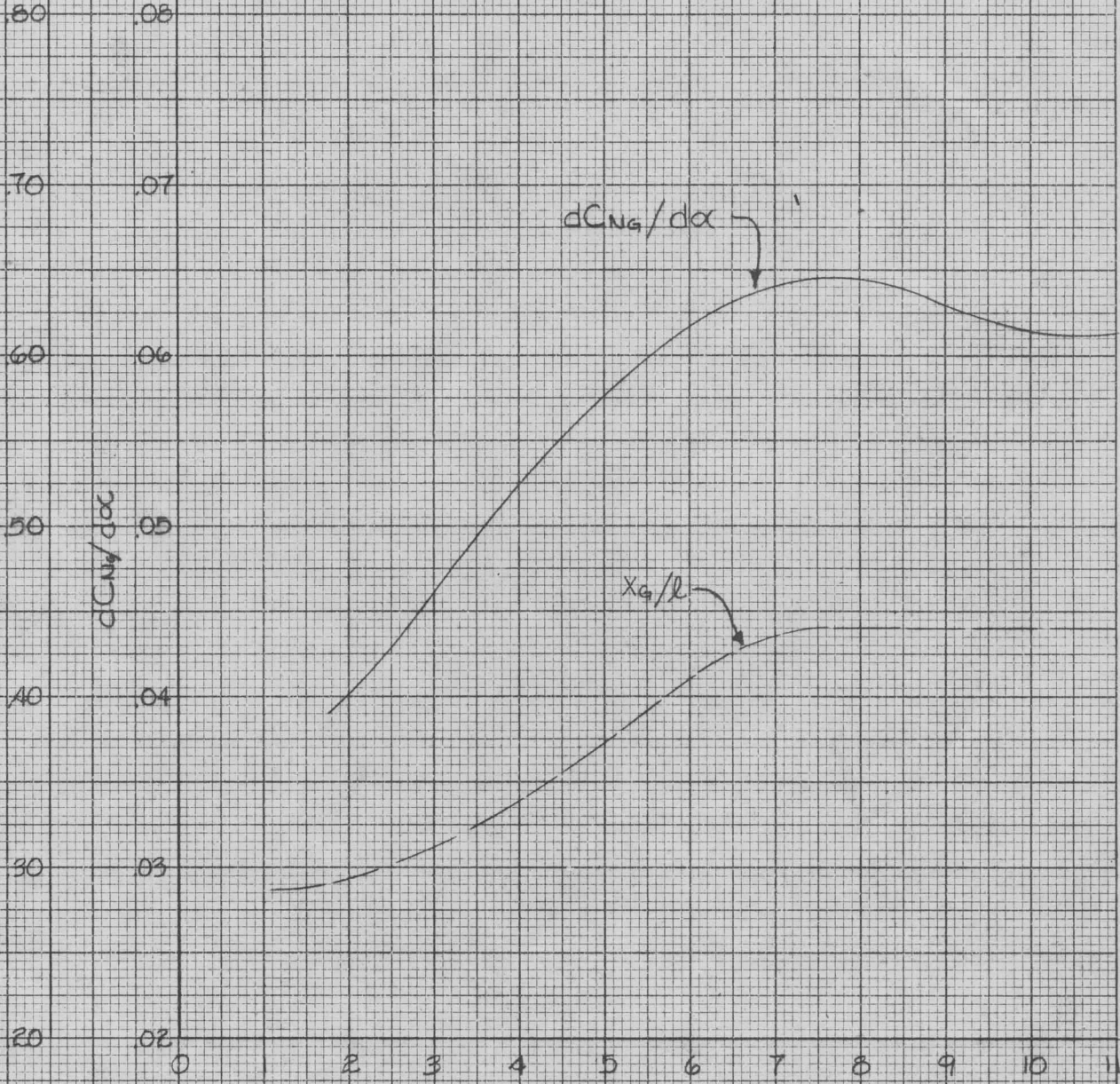
$$C_{Ng} = \frac{N}{q S_B}$$

$$S_B = 15.9 \text{ SQ. FT.}$$

x_g/l = CENTER OF PRESSURE LOCATION AFT OF NOSE / BODY LENGTH

$dC_{Ng}/d\alpha$

MACH NUMBER



NO. 328-11 10 x 10 to the half inch, 2 1/2 inch spacing.
 Engraving, 7 x 10 in.
 MADE IN U.S.A.

KENNEL & ESSER CO.

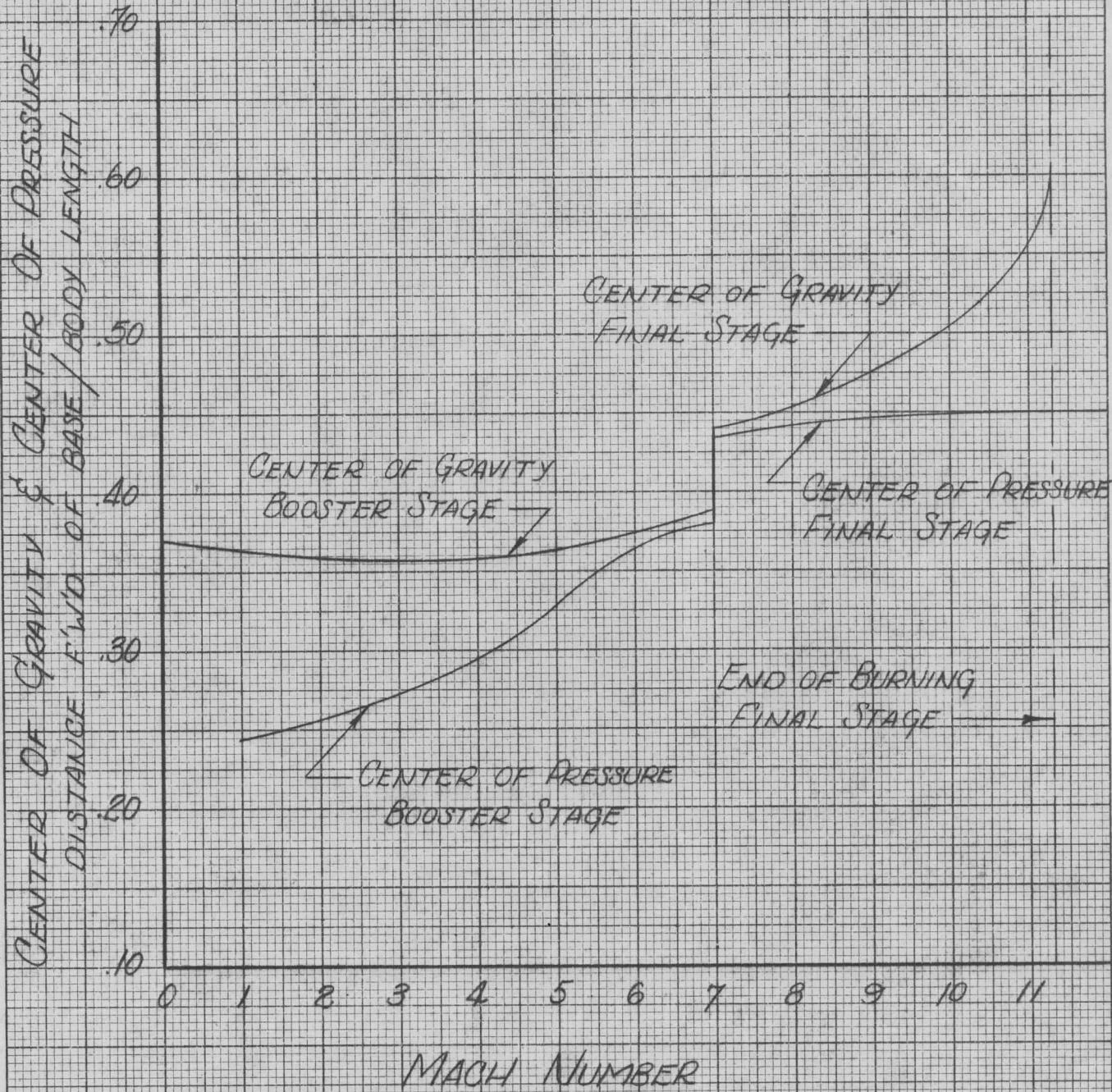
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ROCKET GLIDE TYPE MISSILE

(6000 LB. WARHEAD - 2000 MI.)

VARIATION OF CENTER OF PRESSURE AND CENTER OF GRAVITY LOCATION WITH MACH NUMBER FOR POWERED FLIGHT



NO. 320-11. 10 X 10 to the half inch. 2 1/2 lines secured.
Engraving, 1 X 10 in.
MADE IN U.S.A.

KENNEL & ESSER CO.

SECRET

E. Performance: The 6000 pound warhead, 2000 mile glide type rocket missile is powered by five (5) 20,000 pound thrust rocket motors and four (4) 2000 pound thrust rocket motors. The missile operates in two stages and is unique in the fact that the same fuel tanks are used in both stages, thus saving much air-frame weight. At the end of the booster stage, the four (4) large outside rocket motors are dropped along with related accessories and the entire fin and an appreciable amount of skin and structure. The missile then continues on to the end of burning powered by one 20,000 pound thrust rocket motor and four (4) 2000 pound thrust rocket motors.

At the end of burning of the final stage, the alcohol tank and remaining rocket motors are dropped and thus the remainder of the missile becomes the glide configuration (see page 29).

Powered Flight: A series of step-by-step calculations were conducted in order to obtain the correct altitude and velocity along a horizontal trajectory at the end of burning such that the glide configuration would be in equilibrium at the maximum L/D at this point.

The missile was launched vertically for 20 seconds and then assumed an angle of 77° with respect to the horizontal. The remainder of the flight was along a zero angle of attack path.

The step-by-step calculations were based on the following relationships which are based on the assumption of a zero angle of attack flight path.

$$\frac{dV}{dt} = g \left[\sin \delta + \frac{T - D}{W} \right]$$

$$\frac{d\delta}{dt} = - \frac{g \cos \delta}{V}$$

$$\frac{dh}{dt} = V \sin \delta$$

$$\frac{dR_G}{dt} = V \cos \delta$$

Where:

T = thrust for a given altitude

D = Drag = $q S_B C_D$

W = Weight at a given time

δ^1 = Flight path angle with respect to the horizontal plane at launching site

V = Altitude at a given time

R_G = Range at a given time

t = Time

The variation of thrust with altitude was calculated by increasing the basic sea level thrust of 108,000 pounds for Stage I and 28,000 pounds for Stage II respectively by the percentage indicated in page 18.

The drag analysis used for calculating wave drag and base drag was based on the method outlined in reference 2 for all Mach numbers. These values are identical to those presented in reference 3. The skin friction drag was determined by reference 2 up to M = 2.0 and faired into the values obtained from reference 3 at M = 7.0. Sufficient time did not permit a complete analysis of the method of reference 3 for calculating skin friction and thus the values used at high Mach numbers were taken directly from the RAND report (reference 3) and adjusted to the present glide missile. A plot of the total drag coefficient versus Mach number for the complete powered configuration is presented on page 41 and for the glide configuration on page 42.

Glide Path Flight: The range of the glide path portion of the flight was calculated by the following relationships (see reference 4).

$$g = \frac{V^2}{R} + g \frac{L}{W} \quad (1)$$

$$R_{\text{glide}} = \frac{1}{2g} \int_{U_{E1}}^{U_{E2}} \frac{L}{D} \frac{1}{1 - \frac{V^2}{gR}} dU_E \quad (2)$$

Where: $U_{E1} = V_1^2 + 2gh_1 =$ Initial Energy

$U_{E2} = V_2^2 + 2gh_2 =$ Final Energy

L/D = lift/drag

W = Weight at a given time

δ = Flight path angle with respect to the horizontal plane at launching site

V = Altitude at a given time

R_G = Range at a given time

t = Time

The variation of thrust with altitude was calculated by increasing the basic sea level thrust of 108,000 pounds for Stage I and 28,000 pounds for Stage II respectively by the percentage indicated in page 18.

The drag analysis used for calculating wave drag and base drag was based on the method outlined in reference 2 for all Mach numbers. These values are identical to those presented in reference 3. The skin friction drag was determined by reference 2 up to $M = 2.0$ and faired into the values obtained from reference 3 at $M = 7.0$. Sufficient time did not permit a complete analysis of the method of reference 3 for calculating skin friction and thus the values used at high Mach numbers were taken directly from the RAND report (reference 3) and adjusted to the present glide missile. A plot of the total drag coefficient versus Mach number for the complete powered configuration is presented on page 41 and for the glide configuration on page 42.

Glide Path Flight: The range of the glide path portion of the flight was calculated by the following relationships (see reference 4).

$$g = \frac{V^2}{R} + g \frac{L}{W} \quad (1)$$

$$R_{\text{glide}} = \frac{1}{2g} \int_{U_{E_1}}^{U_{E_f}} \frac{L}{D} \frac{1}{1 - \frac{V^2}{gR}} dU_E \quad (2)$$

Where: $U_{E_1} = V_1^2 + 2gh_1 = \text{Initial Energy}$

$U_{E_f} = V_f^2 + 2gh_f = \text{Final Energy}$

$L/D = \text{lift/drag}$

R = radius of earth

g = gravitational acceleration

W = weight

V = velocity at any time

V_i = velocity at start of glide

V_t = velocity at end of glide

R_{glide} = range of glide path

From equation (1) the required altitude may be determined to attain equilibrium at the angle of attack for maximum L/D using a given weight and velocity at the end of burning.

Equation (2), in turn, will give the range traversed during the glide path portion of the flight.

A plot of the glide path portion of the flight calculated by this method is presented on page 23.

The values of L/D used in the glide path calculations were obtained by

$$L/D = \frac{dC_L}{d\alpha} \alpha}{C_{D0} + \frac{dC_N}{d\alpha} \alpha^2}$$

Where:

$$C_L = C_N \cos \alpha$$

α = angle of attack

C_{D0} = drag at zero angle of attack

The angle of attack chosen for L/D up to M = 7.0 resulted in maximum values of L/D. However, above M = 7.0 values of L/D slightly lower than L/D (maximum) were used in order to operate at a higher angle of attack and therefore higher altitude and lower skin temperatures.

A plot of L/D versus Mach number for the glide flight is presented on page 43.

The re-entry path is considered to be a ballistic path with the glide configuration assuming zero angle of attack.

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PERFORMANCE SUMMARY

	Booster Stage I	Final Stage II
Weight Full	74,775 lbs	17,492 lbs
Weight Empty	20,201 lbs	9,926 lbs
Weight of Fuel (total)	54,574 lbs	7,566 lbs
Thrust at S.L.	108,000 lbs	28,000 lbs
Specific Impulse at S.L.	210 sec	240 sec
Fuel Consumption	514 #/sec	133.3 #/sec
Mass Ratio	3.695	1.762
Initial Acceleration	1.445	1.83
Time of Burning	106 sec	568 sec

	Performance at End of Booster Stage	Performance at End of Final Stage
Velocity Along Flight Path	7049 ft/sec	11,290 ft/sec
Sz - Altitude	87,000 ft	115,000 ft
Sx - Range	190,000 ft	690,000 ft
Flight Path Angle	12.2°	0
Missile Attitude	0	0

Performance at End of Glide

Altitude = 52,000 ft
Velocity = 2,000 ft/sec
Range = 2,005 miles

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M-100-OM CHARLES BRUNING COMPANY, INC. Philadelphia 19th and Locust Streets

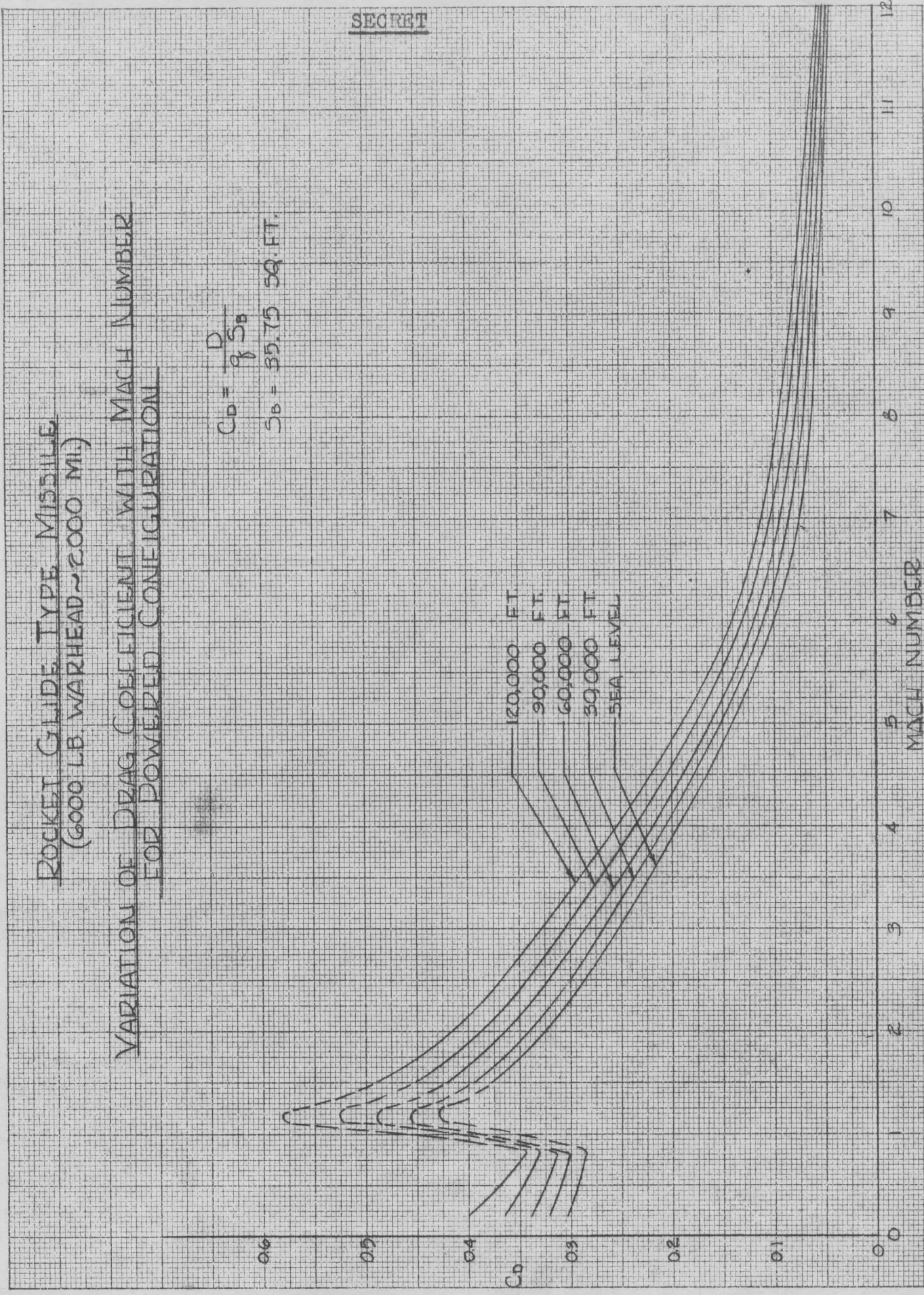
DATA SHEETS

ROCKET GLIDE TYPE MISSILE
 (6000 LB WARHEAD ~ 2000 MI.)
VARIATION OF DRAG COEFFICIENT WITH MACH NUMBER
FOR POWERED CONFIGURATION

$$C_D = \frac{D}{S_B}$$

$$S_B = 35.75 \text{ SQ. FT.}$$

120,000 FT.
 90,000 FT.
 60,000 FT.
 30,000 FT.
 SEA LEVEL

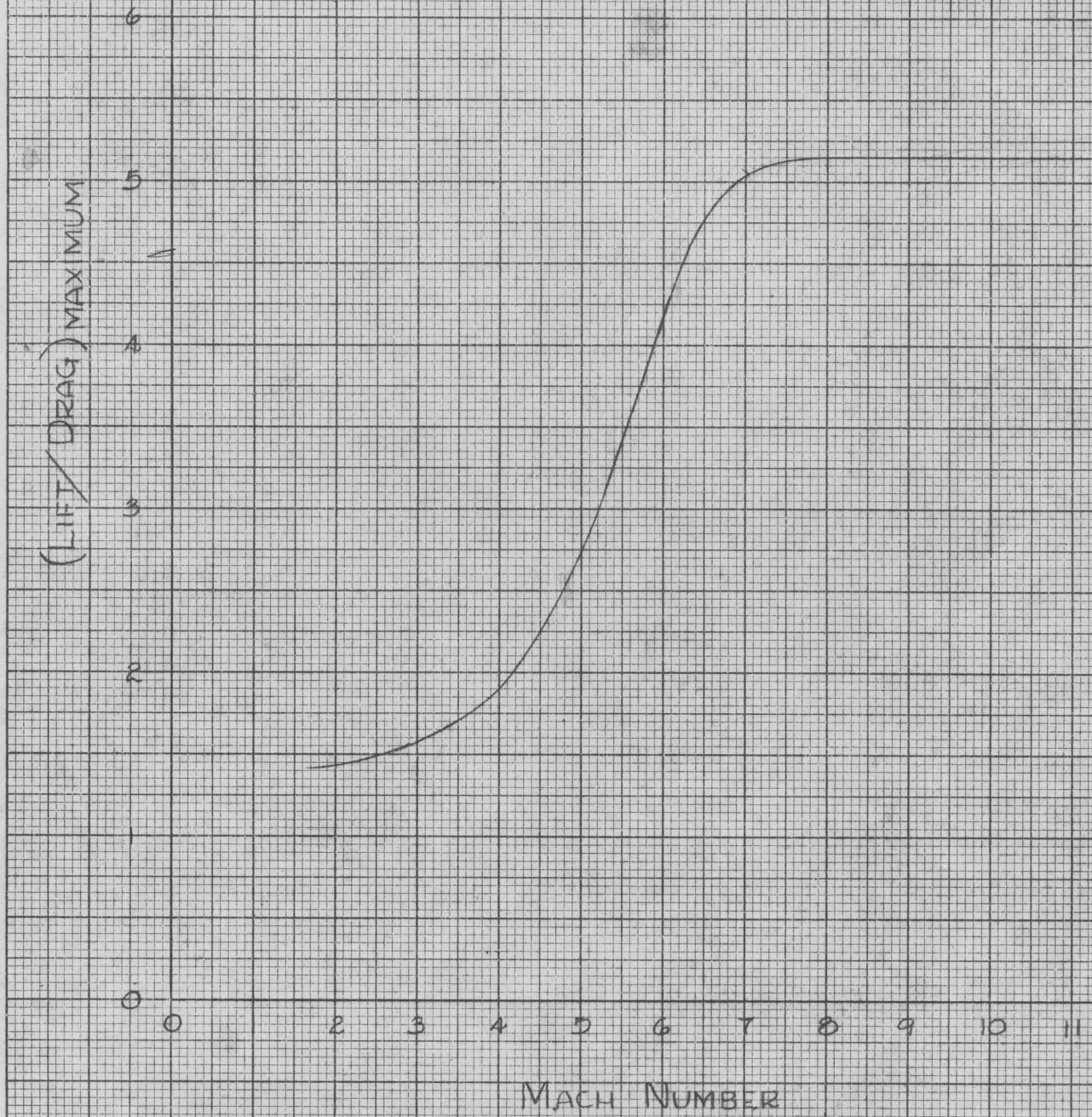


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ROCKET GLIDE TYPE MISSILE
(6000 LB. WARHEAD - 2000 MI.)

VARIATION OF LIFT TO DRAG RATIO WITH
MACH NUMBER ALONG GLIDE PATH



NO. 320-11. 10 x 10 to the part inch. 24th lines secured.
MADE IN U.S.A.
Enlargement 1 x 10 in.

KENNEL & ESSER CO.

SECRET

F. Guidance: The methods of mid-course guidance applicable for a glide path trajectory were presented on page 24 of reference 1. In addition to guidance for the glide path the incorporation of a self contained system of angle of attack control is deemed desirable. Such a system would allow the programming of angle of attack throughout the flight thus permitting (1) flight at best lift to drag ratios at all times, or (2) predetermined control of the flight altitude within wide ranges.

The angle of attack programming signal can come from any one of three sources or combinations of these sources, (1) elapsed time, (2) altitude by positive altimeter, or (3) Mach meter.

It is possible that some disturbance might start the configuration on a skip path. If the angle of attack signal originates from either of the last two sources such a skip would be damped out.

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G. COST ESTIMATE

The following Cost Estimates for the 6,000-pound warhead rocket missile are based on the same cost analysis presented on Page 2.

COST ESTIMATE

6,000 Pound Warhead Rocket Missile

	Unit Cost	Ballistic Warhead 1,000 Mile Range		Glide Warhead 2,000 Mile Range	
		Wt.	Cost	Wt.	Cost
Airframe	\$ 27.00	3,134	\$ 84,500	4,085	\$110,000
Power Plant	80.00	2,110	169,000	2,110	169,000
Guidance	100.00	300	30,000	440	44,000
Warhead	----	6,000	----	6,000	----
Total Wt Empty	----	11,544	\$283,500	12,635	\$323,000
Oxygen	.04	33,852	1,310	33,832	1,310
Alcohol	.10	27,065	2,707	27,065	2,707
H ₂ O ₂	.16	1,243	200	1,243	200
Total Fuel	----	62,140	\$ 4,217	62,140	\$ 4,217
Total	----	73,684	\$287,717	74,775	\$327,217

REFERENCES

1. CVAC Report No. ZP-48-35003, "Proposed Development Program on Rocket Type Missile".
2. CVAC Report No. ZA-6002-002, "Final Aerodynamics Report on MX-774 Single Stage Test Vehicle".
3. Rand Report No. RA-15065, Long-Range Surface-to-Surface Rocket Vehicles Preliminary Investigations and Results. - Aerodynamics
4. Rand Report No. RA-15063, Long-Range Surface-to-Surface Rocket Vehicles Preliminary Investigations and Rockets - Flight Mechanics.